SCIENCE&ENGINEERING SYMPOSIUM

PROCEEDINGS

16-19 OCTOBER

THEME: "Advanced Technologies – Key to Capabilities at Affordable Costs"

VOL. III. FLIGHT DYNAMICS

CO-SPONSORED BY
NAVAL MATERIAL COMMAND
AIR FORCE SYSTEMS COMMAND

- Avionics
- Propulsion
- Flight Dynamics
- Basic Research
- Material
- Armament
- Human Resources

PROCEEDINGS

OF THE

1978 SCIENCE AND ENGINEERING SYMPOSIUM

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NAVAL AMPHIBIOUS BASE
CORONADO, CALIFORNIA

VOLUME III

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PREFACE

The initial co-sponsored Air Force Systems Command/Naval Material Command Science and Engineering Symposium was held at the Naval Amphibious Base, Coronado on 16 - 19 October 1978. The theme of the 1978 Symposium was "Advanced Technologies - Key to Capabilities at Affordable Cost."

The objectives of this first joint Navy/Air Force Science and Engineering Symposium were to:

- Provide a forum for military and civilian laboratory scientific and technical researchers to demonstrate the spectrum and nature of 1978 achievements by their services in the areas of
  - Armament
  - Avionics
  - Basic Research
  - Flight Dynamics
  - Human Resources
  - Materials
  - Propulsion
- Recognize outstanding technical achievement in each of these areas and select the outstanding technical paper within the Navy and the Air Force for 1978
- Assist in placing the future Air Research and Development of both services in correct perspective and to promote the exchange of ideas between the Navy and Air Force Laboratories
- Stress the need for imagination, vision and overall excellence within the technology community, assuring that the air systems of the future will not only be effective but affordable.

Based upon the success of the initial joint symposium (which was heretofore an Air Force event), future symposia are planned with joint Navy/Air Force participation.

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FLIGHT DYNAMICS FUNCTIONAL AREA REVIEW

BY

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Flight Dynamics Functional Area Review

Abstract

This paper presents a brief review of the Flight Dynamics functional area. The primary technologies included in Flight Dynamics are defined as structural mechanics, vehicle equipment, flight control and aeromechanics. The review is presented in two major subsections. The first subsection identifies the principal members of the Flight Dynamics research and development community within this country and Western Europe. Examples of recent accomplishments contributed by the Army, NASA, and U.S. Industry, and Foreign sources are discussed. The second subsection addresses research and development conducted within the Air Force Flight Dynamics Laboratory (AFFDL). Recent accomplishments, technical gaps, and future objectives for each of the primary technologies are identified and discussed. A brief description of the AFFDL's Advanced Fighter Technology Integration Program is also included. The lists of accomplishments presented in this paper testify to the continued advancements being made in the Flight Dynamics area and the lists of technical gaps and future objectives indicate that continued effort is required to ensure that viable aeronautical options are available to counter the expanding technological and operational threat.

FLIGHT DYNAMICS

FUNCTIONAL AREA REVIEW

(FAR)

INTRODUCTION

The process which results in the deployment of new weapons systems begins in the basic sciences and a wide range of technologies where, in response to military requirements, knowledge is transferred into concepts, designs, and experimental hardware; and finally, into weapons systems. When the resulting weapons systems incorporate a flight vehicle, then the process includes those technologies associated with Flight Dynamics. This process is depicted in Figure 1 where aeromechanics, structural mechanics, vehicle equipment, and flight control comprise the technologies generally associated with the Flight Dynamics Laboratory. Within the area of Flight Dynamics, these and the other technologies shown in the figure are integrated.

Within the context of the process which responds to military missions and results in flight vehicles, the objectives of Flight Dynamics can be simply stated as innovate, investigate, and integrate. Flight Dynamics is the topic of this Functional Area Review which includes a brief overview of the total Research and Development Community and rapidly converges to a discussion of the programs conducted within the Air Force Flight Dynamics Laboratory (AFFDL).

The information in this Review appears in two major subsections. The first is an overview of the complete flight dynamics community, defining the members of the United States and European flight dynamics communities. The activities of the members of the flight dynamics community other than the U.S. Air Force and the U.S. Navy are reviewed by identifying and discussing some recently completed or ongoing programs in each area.

The second major subsection reviews the efforts of the Air Force in Flight Dynamics. This section also identifies the FY79 resources in AFFDL and reviews the Laboratory's programs relative to its most outstanding recent accomplishments, the technical gaps it faces, and its future

objectives. The programs, gaps, and objectives are presented in each of four major areas of interest: Structural Mechanics, Vehicle Equipment, Flight Control, and Aeromechanics. This subsection concludes with a short description of the Laboratory's current effort to integrate the technologies of the four major technical areas.

OVERVIEW OF THE FLIGHT DYNAMICS COMMUNITY

The major members of the U.S. Flight Dynamics Community are listed in Figure 2. Most of the organizations listed under the Air Force have contributed at least one paper to this Symposium. The Flight Dynamics Laboratory is the Air Force leader in Flight Dynamics but works closely with, and is highly dependent upon, the other Air Force organizations. The Navy's Flight Dynamics program will be discussed by Mr. Decrescente in the other half of the Functional Area Review.

OTHER GOVERNMENT AGENCIES

The other government agencies which continuously provide significant contributions to Flight Dynamics Technology include NASA and the Army. One of NASA's prime goals is to establish and maintain technological superiority in military aeronautics. The development of advanced technology suitable for future military systems is a major NASA objective. NASA supports joint development with the Department of Defense (DOD) of appropriate research, experimental, and prototype aircraft, to exploit technologies recently developed but not yet sufficiently verified for use in production aircraft and to determine whether attractive mission applications exist. The Army has a defined role in aerodynamics and works closely with NASA in most of its efforts. Examples of recent outstanding programs from these organizations are listed by technology in Figure 3 and are briefly described below.

Aircraft Energy Efficiency Program for Airframes - NASA

Extensive application of advanced composites to commercial transports demands development of a materials data base. This base must be adequate for design to demonstrate weight reduction; to assure durability, maintainability, and repairability; to be competitive in life-cycle costs; and to give management confidence that the product can be de-

livered on time at predictable costs. The Aircraft Energy Efficiency (ACEE) program addresses these needs through contracts with Boeing, Douglas, and Lockheed. Each contractor will develop two composite components that will serve as prototypes for two generic classes of structure. Secondary structure is represented by control surfaces. Primary structure is represented by load-carrying boxes.

The supercritical airfoil with higher aspect ratio and lower sweep represents an aerodynamic innovation likely to find application in all new transport designs. NASA's ACEE, therefore, includes efforts to upgrade aerodynamic-analysis methods and computer codes for the design of supercritical wings. The class of computer codes involving finite-difference approximations will be extended to the full-potential equation for transonic flow past three-dimensional swept wings.

Winglets, another major recent aerodynamic innovation, are configured and positioned so that the effective angle-of-attack of the combined tip vortex and free stream produces a winglet side force with a forward component. Wings with high tip-loading and correspondingly strong tip vortices produce the highest winglet forces. Winglet wind tunnel tests at NASA Langley Research Center included the Boeing KC-135 and Lockheed L-1011. Results from each were compared with the effect of adding a planar wing-tip extension. The winglet compared favorably in each case, but showed best on the KC-135 because of its high tip-loading. Active controls can alleviate wing loads by rapid deflection of the ailerons to redistribute the load inboard. Flight-test results for two different maneuvers of the L-1011 system show that bending moments, which doubled with the system off, were held to 1-g or below, with the system on. The earlier applications of active controls do not critically affect safety of flight. Use of active controls in flight-critical applications to gain maximum performance benefits will be studied in depth.

Of all possible technology advances, maintenance of a stable laminar boundary has the greatest potential for reducing fuel consumption. Laminar Flow Control (LFC) over 75% of the wing and 65% of the empennage of a 5500-nautical-mile, 200-passenger transport could reduce direct operating costs (DOC) about 10%. In view of such potential, NASA included LFC in the ACEE program. Compatibility of LFC with advanced supercritical wing geometries is being defined and analytically evaluated.

The major ACEE effort on LFC consists of a system-definition contract by Boeing, Douglas, and Lockheed. One contractor prefers bonded metal honeycomb with slotted surfaces and integral ducts. Another integrates the ducting system in a stiffened graphite-composite substructure with a bonded titanium outer surface with slots. The third has fiberglass glove panels with collecting ducts and a woven stainless steel porous surface; the glove panels overlay an externally stiffened composite substructure providing additional ducting.

Standard Modular Cooler - Army

The Army Standard Modular Cooler is a Stirling cycle cooler that is highly optimized for maximum thermal efficiency and light weight. The heat transfer surfaces are designed with fins to increase the heat transfer area and are constructed of materials selected to provide high thermal conductivity in order to obtain high thermal efficiency. The design minimizes both the temperature rise due to the compression process and the heat leak into the cold end of the cooler, resulting in lower input power requirements. The cooler uses a brushless DC motor. The drive motor and all moving parts are designed for quiet operation so as to be undetectable in a battlefield situation.

The cooler was built to be compatible with several man-portable and vehicle-mounted infrared night sights and has undergone extensive manufacturing methods development.

Structural Mode Stabilization, YF-12 - NASA

NASA is pursuing a program for implementing small forward control surfaces on the YF-12 to stabilize several low frequency longitudinal structural modes. The significance of this program is the demonstration and validation of active structural mode stabilization techniques at high supersonic Mach numbers where data are now nonexistent.

Considerable design data and information exist from the B-52 Load Alleviation Mode Stabilization (LAMS) program, the C-5 Active Load Distribution Control System (ALDCS) program, and the B-1 aircraft; however, validation and correlation with flight test have been primarily for subsonic conditions. This program represents the only capability to demonstrate the concept at high Mach number and altitude and to provide data for design purposes.

Simulation Technology for Rotorcraft in the Nap-of-the-Earth Environment - Army

The Army sponsored a series of efforts for the development of wide angle image generators and display techniques. These techniques have the potential to provide sufficient detail for Nap-of-the-Earth (NOE) flying. The program studied the trade-off between computer capability and pilot cueing requirements and determined the performance required to provide high-fidelity cueing during NOE maneuvers. The program also assessed the requirements for real-time solutions of rotorcraft math models and developed a hybrid analog/digital special purpose computer that can perform rapid solution of complex models. These results enhance training simulator technologies and provide improved simulator capabilities for advanced helicopter systems integration.

Non-Axisymmetric Nozzle/Airframe Integration - NASA

NASA conducted several investigations examining the benefits associated with non-axisymmetric nozzles, with in-flight thrust vectoring and/or reversing capabilities. NASA Langley Research Center, for example, began by testing vanes placed in an exhaust flow, and then proceeded to investigate a variety of 2-D nozzle configurations on an isolated pod model. They subsequently have tested several 2-D nozzle configurations on 1/12-scale F-15 and 1/10-scale YF-17 jet effects models to determine what performance and maneuver benefits may be accrued over a baseline aircraft. Langley has also conducted one-on-one manned simulations of a baseline F-15 versus an F-15 with 2-D nozzles and canards. NASA Lewis Research Center has undertaken investigations of the cooling requirements and the development of higher aspect ratio 2-D nozzles. NASA Dryden Research Center has supported a YF-17/2-D nozzle systems study and conducted a 2-D nozzle technology workshop for industry and government agencies.

This coordinated effort has been directed toward improving the data base in the disciplines of propulsion, aerodynamics, structures, and vehicle integration. The effort expended by the NASA centers has been of significant value to the Air Force in exploring new technologies for future aircraft applications.

INDUSTRY INDEPENDENT RESEARCH AND DEVELOPMENT

The Aerospace Industry, Universities, and Research Centers do much of the Flight Dynamics work, most of which is supported by the government. A significant amount of effort is conducted under the Industry's Independent Research and Development (IR&D) program. The total IR&D 1977 budget was greater than $1.5 billion with approximately $260 million directed towards Flight Dynamics technologies. These programs are reviewed and evaluated by members of the government Flight Dynamics Community. A few of the most significant IR&D programs are listed by technology area in Figure 4 and are discussed below.

Metal Matrix Composites - Rockwell

Recent advances in fiber technology have resulted in several fibers whose properties include very high strength and stiffness. Some of those fibers, e.g., boron and graphite, have been combined with organic matrices to form composite materials. Those composite materials have certain limitations to their usage, particularly due to the effects of elevated temperatures on the properties of the matrix. The combinations of the advanced fibers with various metals as the matrix material appear to be viable structural candidates. The various fiber/matrix combinations provide a wide range of strength, weight, and temperature characteristics. The Rockwell Los Angeles Division has a very active IR&D program to assess some of the more attractive combinations that utilize powdered titanium or aluminum, or titanium foil as the matrix, combined with silicon carbide or boron fibers. The diffusion bonding characteristics of titanium, as well as its superplasticity, make for an ideal matrix material, providing the ability to form a fiber/matrix foil which may be used to achieve the desired strength and stiffness required by that structural component.

The use of the advanced fibers in combination with a metal matrix provides potential weight savings of up to 60% over conventional titanium structure. In addition, depending on the percentage of fiber utilized, acquisition cost savings of 15-25% can be achieved over a conventional titanium structure design.

Active Flutter Suppression - General Dynamics/Lockheed

General Dynamics has an IR&D program to develop and

demonstrate a flutter suppression system (FSS) for the F-16 aircraft which uses existing control surfaces to improve the flutter characteristics of the aircraft when carrying external stores. A design study was conducted for one external store take-off loading. The study verified that the flaperon could suppress flutter for this store loading. A hydraulic system for the flutter model was designed and fabricated. Bench tests of the hydraulic system were conducted to provide basic information on the system dynamic characteristics. An existing 1/4-scale flutter model has been redesigned to accommodate the flutter suppression system. Wind tunnel tests are planned for FY79 to demonstrate the FSS on the F-15 flutter model.

The Lockheed-California Company is adding active controls to the L-1011 to make it possible to increase the wing span, and thus reduce fuel consumption, without significant structural change to the original wing. The active control functions that are being added are wing load alleviation using maneuver load control, gust alleviation, elastic mode suppression, and flutter margin augmentation. For the maneuver load control function, an accelerometer near the airplane's center of gravity senses when the load factor is greater than 1 ($n>1$). Both outboard ailerons are rotated, trailing edge up, to cause a modification of the spanwise lift distribution such that the center of pressure is more inboard than for the normal distribution, resulting in a reduction of the wing root-bending moment. Alleviation of gust loads associated with aircraft pitch response is accomplished by stabilizer rotation in response to fuselage rate of pitch. For elastic mode suppression, a wing tip accelerometer senses wing bending and the outboard aileron is rotated symmetrically to damp the wing bending response. When elastic mode suppression is used, the techniques of flutter augmentation may be required to prevent unfavorable effects on the flutter stability. Flutter margin augmentation is aimed at maintaining flutter speed and damping margins equivalent to those of the basic airplane. Lockheed's work in active controls started as in-house R&D in 1974. A considerable amount of analytical, ground test, and flight test work related to the active control concepts is now being funded by NASA as part of NASA's Aircraft Energy Efficiency (ACEE) Program.

Response of Composite Structures to Ballistic Impact - Boeing

With the increased usage of advanced fiber-composites as a structural material in aircraft components, understanding

the effects of nonnuclear threat-induced damage mechanisms in a combat environment becomes increasingly important. Aircraft structural designs using fiber-composite materials present some unique problems, one of which is the sensitivity of structural fiber-composite materials to physical damage. Another problem is the low energy absorption ability of structural fiber-composite materials. Moreover, structures made from fiber-composite materials tend to have few discontinuities because of the inherent weight penalty for effective transfer of load between panels. Consequently, such designs have few built-in crack stoppers for damage arrestment. This, plus the tendency for extensive damage resulting from dynamic pressure pulses, warrants the investigation of nonnuclear threat-induced damage mechanisms as well as the effect this damage has on the strength of the structures.

Once the damage processes are more fully understood, not only can the associated strength reductions be more accurately predicted, but also the designs can be tailored to be more battle-damage tolerant. The Boeing Company has initiated a program (See Figure 5) to develop and demonstrate the technology required to (1) predict specific levels of damage resulting from nonnuclear threat-induced damage mechanisms on aircraft structural fiber composite materials, (2) relate specific levels of damage with reduction in aircraft structural strength, and (3) design and demonstrate fiber-composite structures that are highly tolerant to severe levels of battle damage.

Cryogenic Radiator - Rockwell

Reliable cooling methods must be developed for large infrared detector arrays planned for use on Air Force satellites. One approach is direct, passive radiation to space. Rockwell International designed, fabricated and tested a fifteen-square-foot feasibility model of a two-stage radiator under an IR&D program. Test data verified analytical predictions and established the effectiveness of the design.

The radiator design approach developed by Rockwell is simpler than other approaches because specularly reflecting deployed sun shields are not required. The concept was selected for full-scale development of a test model radiator under an AFFDL (SAMSO funded) contract. The full-scale model will have a projected area of 103 square feet with a cooling

capability of five watts at 70°K. This is fifty times the cooling capability of cryogenic radiators presently used on satellites and will operate at a temperature 10 to 40°K lower.

Redundancy Management - Boeing/Lockheed

IR&D programs are underway at the Boeing and Lockheed Companies examining the application of observability to redundancy management in order to reduce the equipment complement. Reliability of military aircraft control systems is not maximized, nor are costs minimized, with current equipment redundancy management techniques. The problems of failure detection, fault isolation, and system reconfiguration around the fault must be covered with probabilities approaching 100% to assure adequate system integrity. Software capability permits the application of observers for more advanced systems using digital technology. The combined hardware and software system can achieve the equivalent capability of a triplex system with a dual-redundant equipment complex.

F-16 Variable Geometry Inlet - General Dynamics

The objective behind the existing F-16 pitot inlet was to provide a lightweight, low cost, simple system that had excellent transonic maneuvering characteristics. In developing the F-16, however, General Dynamics designed the basic airframe with higher Mach number capability in mind. Specifically with respect to the inlet, the front portion was constructed of fiberglass and made easily detachable. Thus, little or no change to the basic vehicle structure would be required to incorporate another type of inlet.

Under an IR&D program, General Dynamics developed a simple variable geometry inlet. Aero-force and inlet testing were accomplished to define the stability, performance, and inlet distortion characteristics of the configuration. A detailed design was undertaken to refine the structure, mechanical design, and weight of the variable geometry inlet.

The use of a variable geometry inlet in place of the pitot inlet significantly improves the supersonic capability of the F-16. For example, inlet total pressure recovery is increased by more than twenty percent at Mach 2.0. An improvement of this magnitude offers a substantial increase in specific excess power and sustained turn capability.

This is accomplished with little or no penalty in either performance or inlet-engine compatibility characteristics at transonic maneuvering conditions.

Advanced Combat Aircraft Concepts - Boeing

The objective of this effort is to identify and evaluate new combat aircraft concepts for tactical air warfare applications. Conceptual systems such as supercruise fighters, low-cost fighters, STOL fighters, and laser carriers are being developed; initial technology requirements are being defined and preliminary program plans and schedules are being developed. A major part of this IR&D effort is to assess the impact of advanced technologies on these concepts and to compile a new data base of weight and cost factors.

Under this program, Boeing has evaluated seven supercruise designs and selected an arrow wing concept for detailed studies. Included in the in-depth studies were high altitude life support systems, handling qualities, aeroelastic design methods, and alternative airframe designs. These areas are considered critical to the development of a successful supercruise fighter. Also included in the IR&D program were low-cost options for advanced technology configurations that achieve strategic weapon delivery, two-dimensional nozzle design studies, CONUS defense, and supercruise reconnaissance-strike studies. Current plans call for evaluation of propulsion/flight control coupling with two-dimensional exhaust nozzle thrust vector control at all flight conditions including stall spin recovery, detailed conceptual design of the supercruise baseline, and an evaluation of mission/system design integration.

Through the organization of this IR&D effort, early identification and accurate assessment of advanced technologies and concepts is possible. It helps provide the detailed information necessary to embark with confidence on technology development programs.

EUROPEAN FLIGHT DYNAMICS COMMUNITY

In addition to the U.S. Flight Dynamics Community, a European Flight Dynamics Community exists which is about half the size of its U.S. counterpart. The principal members of the European Community are listed in Figure 6. European aerospace technology has produced a large number of significant advances. Some of these are listed by technology

area in Figure 7 and are described below.

Active Flutter Suppression - France

The French National Institute for Aerospace Studies and Research (ONERA) is rapidly progressing in the development of active flutter suppression technology for application to fighters to obviate the need for speed placards when external stores are carried. They completed wind tunnel tests on a research wing several years ago and are currently testing an F-1 flutter model with a trailing edge active flutter sppression system. Two different store configurations have already been tested and evaluated. The improvements in allowable flutter speed were impressive. Their next planned step is to develop a digital control system for the model which will be adaptive from the standpoint of Mach number variations. This should be accomplished within the next year and should be ready for test shortly afterward. Also, analyses are progressing by the French on applying their flutter suppression approach to an AFFDL-developed flutter wind tunnel model. The AFFDL will conduct the model tests to evaluate this application of the French-developed control laws. Control laws provided by the United Kingdom British Aircraft Company/Royal Aeronautical Establishments and Israel (Dr. Nessim under NASA/LRD Grant) will also be evaluated with the model.

Active Flutter Suppression - Germany

The Germans are also very aggressively pursuing the development of active suppression concepts for wing store flutter. Germany is probably now the leader in Europe, following the completion of a detailed model test program and a flight test demonstration on the Fiat G-91. Through a joint Memorandum of Understanding between the AFFDL and the FRG Ministry of Defense, active store flutter suppression will be demonstrated on an F-4 aircraft using a conventional trailing edge aileron as the active surface. The flight tests for the program will begin in November 1978 and will continue for about three months. The Federal Republic's Aerodynamic Research Institute (AVA) is also working with Messerschmidt-Boelkow-Blohm (MBB) to develop a facility for determining impedance transfer functions for hydraulic actuators, with preload, over a wide frequency range. The F-4 actuators to be used in the early flutter suppression flight have been tested at the AVA facility. AVA is also responsible for analyzing and evaluating the unsteady aero-

dynamics to be measured on the F-4 ailerons during the flight tests. Next year, AVA will test an MBB flutter suppression model in their wind tunnel.

Active Flutter Suppression - United Kingdom

The work completed in the UK in the last several years emphasized theoretical development. They have very few wind tunnels in the UK suitable for active flutter suppression tests. Through the Royal Aircraft Establishment, the British Aeronautical Company is conducting analyses to obtain control laws for the AFFDL flutter suppression model, using their design philosophy. These control laws will then be tested by AFFDL using the AFFDL flutter model in the NASA Langley Research Center 16-foot wind tunnel. The UK are now defining plans to conduct subsonic model tests in 1979 to explore explosive type flutter.

The activity in the European Community is clear indication that they are rapidly achieving a state of the art where they will be applying this technology to near-term aircraft systems. It is imperative that the U.S., particularly the USAF, continue to vigorously pursue this technology for our own weapon system applications if we are to maintain our "technological edge" in combat weapon systems.

Powder Packs for Aircraft Fire Protection - United Kingdom

Current techniques utilize external foam to provide dry bay fire protection against ballistic combat threats. Foam, however, imposes severe maintenance, accessibility, and environmental penalties. Chemical powder packs provide a viable alternative of high potential merit. The powder pack concept is based on dry chemical powder (extinguishing agent) sandwiched between two thin sheets of suitable material. The powder pack is bonded to the entire area of fuel tank skin inside the aircraft. Projectile penetration and blast will disrupt the sandwich construction and disperse the extinguishing agent even prior to fuel leaking from the tank. The powder pack has the potential of using any suitable dry chemical agent, such as Purple K. So far, the Royal Aircraft Establishment has been the primary developer of the powder pack concept, investigating the potential of using MONEX 751 powder (trade name). It is believed that this powder type is the most effective extinguishing agent currently available. MONEX 751 is

basically Purple K (potassium bicarbonate, $KHCO_3$) with a urea additive. There are several other agents presently on the market; however, the exact mechanism for fire suppression is, as yet, not fully understood. Three theories have been advanced to explain the mechanism: (1) The dispersed powder provides a large surface area for free radical recombination which would then interrupt the combustion reaction; (2) Since all extinguishing agents are basically "salts," the ions in the powder will actually react with the free radicals to interrupt the combustion reaction; and (3) There is a significant heat transfer to the powder which tends to quench the reaction. Any or a combination of these theories may prove to be true and additional analyses are currently being performed.

Application of Fly-By-Wire Techniques To the Mirage 2000 - France

The Mirage 2000 is a production aircraft employing advanced flight control technologies. It is equipped with a fly-by-wire (FBW) system with no mechanical backup. This gives the system quadreplex in roll and pitch with a triplex configuration on rudder. The prime consideration for installation of the fly-by-wire system is capability to relax static stability for increased maneuvering performance, increased flight safety at the flight boundries, and a capability for mission-oriented controls and growth potential.

Application of Control Configured Vehicle and Fly-By-Wire Technology to Concorde - France

The production Concorde primary flight control system includes a three-channel FBW command signalling scheme in conjunction with a normally de-clutched mechanical backup. This design philosophy is based on the facts that a mechanical flight control system alone cannot provide the necessary flying qualities due to kinetic heating of the mechanical linkages, long cable/rod runs, and the wide variation in aircraft aeroelastic properties.

As a part of a continuing SST technology development program, the Number 1 production Concorde was recently modified and flown with a 4-channel digital flight control system and sidestick controller. The flight test program was to explore performance improvements using negative stability CCV concepts. The purpose of the tests was to examine technology that could be applied to future civil and military transport aircraft and possibly to combat aircraft as well.

Application of Fly-By-Wire to KFIR - Israel

The production KFIR incorporates a full-authority FBW primary flight control system to enhance mission related flying qualities and also improve survivability through dispersed redundancy. A follow-on version of the basic KFIR is currently being evaluated. This configuration designated KFIR C-2 incorporates swept canards mounted on the upper portion of the intakes. The canards in conjunction with other aerodynamic changes allow greater exploitation of the basic FBW system by providing improved maneuverability and flying qualities.

Future German Combat Fighter - FRG

Among several outstanding European research and design efforts on NATO fighter aircraft, is the program by Messerschmidt-Boelkow-Blohm (MBB) of the Federal Republic of Germany. MBB has been conducting preliminary design studies and some component tests on an advanced delta-wing/canard fighter. This is primarily for the air-to-air combat role to augment the MRCA multi-role fighter. Some information on the design features has been shown in various German aircraft publications and AGARD reports.

The MBB design incorporates advanced aerodynamic concepts such as relaxed static stability/fly-by-wire and close-coupled canards for high maneuverability. Advanced flight modes, including direct lift/direct side-force, and drag modulation are being evaluated by analysis and manned flight simulation. Additionally, MBB is evaluating a flight mode called post-stall control technology (PST) which allows the aircraft to operate at very high angle-of-attack in a stable manner at low flight speeds, where the conventional aerodynamic surfaces are not highly effective. The aerodynamic control forces in this flight regime are augmented by propulsion forces through thrust-vectored exhaust nozzles. The utilization of this flight regime in close-in air-to-air combat with missiles and guns is being analyzed and evaluated in manned simulations by MBB and the FRG Ministry of Defense.

The outstanding characteristic of this research is the integration by MBB of various aerodynamic and propulsion technologies with fairly detailed analyses and simulations to determine effectiveness and utility. This is an approach used quite heavily in various AFFDL programs and perhaps the US and FRG technical specialists can learn from each

other in an effort to reduce research costs.

Flight Dynamics Work in the USSR

The final major member of the world Flight Dynamics Community is the USSR. Little will be said in this presentation regarding specific USSR Flight Dynamics programs other than that the USSR is active in nearly all areas of Flight Dynamics as evidenced by such recent developments as the Backfire and Fencer aircraft. Flight Dynamics developments in the USSR are closely watched and evaluated in this country and one of the papers in this session addresses Soviet capabilities.

AIR FORCE FLIGHT DYMAMICS LABORATORY

The Air Force Flight Dyamics Laboratory (AFFDL) is part of the Air Force Wright Aeronautical Laboratories (AFWAL), a four-laboratory organization located at Wright-Patterson AFB, Ohio. The AFFDL has unique research and technology responsibilities for flight vehicle structural mechanics, vehicle equipment, flight control and aero-mechanics. The AFFDL provides direct technical support to weapon systems development programs and other Laboratories and Services in these areas. The Laboratory's programs are, of course, influenced by the technologies developed by other Air Force Laboratories, e.g., Materials, Weapons, Propulsion, Avionics, and Human Resources. The maximum beneficial influence on flight vehicle performance and mission effectiveness is obtained through the integration of all the developed technologies and it is the AFFDL which performs this function within the Air Force. Analytical programs are continually conducted within AFFDL to evaluate the effects of technologies and combinations of technologies on flight vehicle performance and mission effectiveness. Technology efforts within the AFFDL range from the fundamental elements of research, through innovative exploratory development and advanced development, to demonstration.

For those disciplines included in Flight Dynamics, the Laboratory is responsible for developing a strong technology base which provides the technological options for the development of Air Force systems and prevents the possibility of technological surprise from potential adversaries. The major thrusts in the AFFDL are: to enhance effectiveness; to improve survivability; to reduce operation, maintenance and acquisition costs; and to provide improved design criteria and capabilities for simulation and analysis. These thrusts are very much in line with the theme of this year's S&E Symposium "Advanced Technologies - Key to Capabilities at Affordable Costs."

In FY79, the Laboratory will expend $106.1 million in carrying out its mission. Figure 8 shows the source and distribution of the Laboratory's funds. Of the $83 million of Laboratory funds, $49.7 million is for exploratory development programs, $29.3 million is for advanced development programs and $4.0 million is for basic research. The funds associated with each discipline are also shown in

the figure. For any given year, the funds distribution would not be expected to vary too far from equal amounts for each discipline. In the following sections, each of the four Flight Dynamics disciplines will be briefly reviewed relative to their individual objectives, goals, and areas of emphasis. These reviews are followed by a brief description of the Laboratory's efforts to integrate the technology from the four major areas.

STRUCTURAL MECHANICS

The objective of the Laboratory effort is Structural Mechanics is to provide demonstrated technology which will result in lower cost, lower weight, improved performance, and an assured design life for advanced flight vehicles. In addition to building a strong structural mechanics technological base and providing technical support to other Air Force organizations, specific goals which are being pursued include: the demonstration of the readiness and advantages of advanced design composite and metallic airframe structure; the enhancement of the structural integrity and environmental durability of metallic and composite structures; and the validation of low-cost innovative structures and aeroelastic concepts.

Structural mechanics programs and concerns are listed in Figure 9 in terms of recent accomplishments, technical gaps, and significant future objectives. These programs are representative of the structural mechanics efforts in AFFDL. Each of the programs listed in the figure is briefly described below.

Completion of Initial Durability Verification of Primary Adhesively Bonded Structure

The development of primary adhesively bonded structure (PABST) technology is being supported by the AFFDL. Through the use of adhesive bonding, large cargo aircraft structures such as fuselages can be reduced from the labor-intensive assembly of thousands of parts to the assembly of a few large bonded modules.

Adhesive bonding is not a new technology. It has been used for more than thirty years in aircraft. Its applications have been limited, however, to secondary structure. Service experience has been quite varied,

ranging from excellent to very marginal. The problems have been random, premature bondline corrosion and delamination, both in metal laminate and honeycomb structure. In the late 60s, intense studies of these failures produced an unprecedented level of understanding of the intricacies of an adhesive bond. This produced the latest state of the art of materials, processing, and quality assurance technology which holds the promise for long-life high-durability bonded aircraft structures.

Fatigue cracking and joint faying surface corrosion are frequent problems facing the Operational and Logistics Commands; the former arising from the stress concentration effects of rivet holes, and the latter due to moisture entry around fasteners. Adhesive bonding, used in primary safety-of-flight critical structure in lieu of mechanical fasteners, can greatly reduce these acquisition and life cycle cost burdens. PABST objectives are to demonstrate improved structural durability while achieving a twenty-five percent acquisition cost reduction and fifteen percent weight reduction through the use of adhesive bonding in primary cargo fuselage structure.

The step up from secondary to primary structure is a big one. Structural integrity and durability must be demonstrated in all of its aspects before confidence in the technology is assured. PABST is that critical demonstration, raising adhesive bonding from the artsmanship of secondary structures to a well characterized, well understood engineering discipline for primary aircraft structure. PABST, initiated in February 1975, is a four-year, four-phase program. The phases are Preliminary Design, Detail Design, Manufacturing, and Fatigue testing of the adhesive bonded 42-foot long, 18-foot diameter fuselage section of the YC-15. Having passed all critical milestones, including the limit proof pressure test, full scale fatigue testing is now underway. Two lives (38,000 pressure cycles) were successfully reached on 12 September 1978, with the goal being four equivalent lifetimes.

<u>Advanced Composite Horizontal Tail, B-1</u>

The AFFDL recently completed the development of an advanced composite horizontal stabilizer for application to the B-1 aircraft. This horizontal tail is the largest advanced composite structure developed to date. The stabilizer is approximately 240 square feet in area and

pivots on a shaft that extends from the vertical stabilizer. The torque box consists of spars, ribs, covers, and the bearing support fitting. The materials systems employed in the design include 3501/AS graphite epoxy and 5505 boron epoxy. It is mechanically fastened using conventional aircraft fasteners. Over one thousand coupons and elements were tested, as well as a multitude of subcomponents, to study basic strength, lightning strike, thermal pulse, acoustic fatigue, moisture, foreign object damage, effects of damage tolerance, etc. Four full-scale articles were completed for static test, fatigue test, and flight test. The structure achieved major cost and weight savings in comparison to its metal counterpart. The composite structure was 14 percent lighter than its metal counterpart and saved 17 percent in cost. These weight, and especially the cost, savings resulted in its selection for planned production for the B-1 aircraft.

Effects of Fastener Hole Quality on Structural Life

In studying the quality of production fastener holes in primary structural elements, it was discovered that microscopic scratches were the primary cause of short structural lifetime. The damaging scratches had been previously overlooked since they are smaller than can be detected by the best existing production line nondestructive inspection/nondestructive evaluation (NDI/NDE) techniques. The source of these scratches was found to be a nonrotating drill during the retraction phase of drill operation. Some relatively simple modifications were made to the drill motor so that the drill continued to turn throughout the retraction. The result was a decrease in microscopic scratches and a significant increase in structural lifetime. With only this modification to the drill motor, it was determined that statistically 98% of those structures drilled with the modifications gave life performance better than the mean of those made with current production methods. Additional changes to production methods have given even better results.

Because of the obvious improvements, the drill mechanism modifications have been incorporated into production lines of three major airframers, General Dynamics/Fort Worth, Lockheed/Georgia and AVCO. There is considerable interest from Air Force Logistics Command (AFLC) centers where hole rework is accomplished to incorporate these modifications.

Further manufacturing-related benefits, such as task elimination and change in assembly procedures, were identified and will be pursued under an Air Force Man-Tech program for the F-16 Systems Program Office. Minimum potential savings are $20,000 per airframe for the F-16 (over 3,000 F-16s are projected). The potential reduction in cracking problems directly translates into an immense reduction in cost of ownership for inspection and rework under AFLC efforts.

Design Manufacturing Methods For Elevated Temperature

Conceptual design studies of the next generation cruise missile identified candidate designs which cruise at high Mach number. These studies also showed that increased missile penetration distance and effectiveness were achieved through reduced Radar Cross Section (RCS).

New radar absorbing structural concepts are available which are load carrying and offer the promise of significant weight savings over non-load carrying radar absorbing materials. These concepts are called Radar Absorbing Primary Structure (RAPS). Much work remains to demonstrate that these new approaches are capable of withstanding the operating temperatures and environments of the next generation cruise missile. Efforts are required for the design, fabrication, and test of structural components made of load bearing materials which absorb or dissipate radar signals and operate in a high temperature environment.

Computational Methods For Transonic Unsteady Aerodynamics

Recent progress under the AFFDL basic research program has shown the aircraft designer how to perform future analyses for unsteady aerodynamic loads, flutter, and stability. Up to now, no transonic aerodynamic methods were available because of the mathematical nonlinearities in transonic flow. This year the AFFDL succeeded in performing a flutter analysis of a simple conceptual airfoil using simultaneous time-integration of the structural and nonlinear aerodynamic equations. The flutter stability was found to depend on the magnitude of the initial disturbance. This was contrary to our previous experience with linear systems. The AFFDL has recently initiated an effort to convert these successful research results into production computer programs for realistic

solution of aircraft design problems in loads, gust-response, flutter, and aircraft stability and control.

Computer Aided Design Techniques

In recognition of the importance of interactive graphics to structural analysis methods, the AFFDL has started to develop a generalized integrated graphics structural analysis system called STAGING (STructural Analysis via Generalized INteractive Graphics). STAGING is a highly interactive system intended to synthesize the finite element methodology into a cohesive, user-oriented capability and radically reduce the time required to conduct a finite element analysis. The system allows users to rapidly generate finite element models and interpret analysis results independent of the analysis program chosen to conduct the analysis.

STAGING consists of six major modules: Executive Monitor, Preprocessor, Display and Edit, Postprocessor, Analysis Programs, and Generalized Data Base. The Executive Monitor helps a user find and use a particular capability and ensures the proper flow of information between modules. The Preprocessor is used to generate bulk information for the analysis codes. The Display and Edit package provides a host of interactive graphic utilities to assist in "fine tuning" previously generated data and effectively displays analysis results. The Postprocessor allows easy generation of additional engineering information from the basic output files of the analysis codes. The Analysis Module consists of available design and analysis computer programs. Finally, the Generalized Data Base provides efficient storage for all geometric and non-geometric information associated with a particular analysis.

Currently, the Level II version of STAGING has been received from the contractor and the program is being made operational on the Aeronautical Systems Division Computer System. STAGING is a program designed for the future because of its modular nature and emphasis on interactive graphics and a real time environment. The development of the STAGING system is expected to continue past the current development phase. Some future possibilities include a minicomputer version, voice-activated menus, and color graphics.

Strength and Durability Requirements For Advanced Composite Primary Structures

Current Air Force structural integrity policy requires that all "safety of flight" structure on all new Air Force aircraft will be designed to be damage tolerant; that is, it requires that the structure have a specified minimum residual strength at the end of a specified period of unrepaired service usage. Compliance with this policy requires the ability to predict failure modes, residual strength levels in the presence of probable types of material and/or manufacturing deficiencies and/or service-induced damage, and the degradation in residual strength as a function of "real time" operational usage.

The Air Force structural integrity policy also requires that all new Air Force aircraft be designed to be durable; that is, it requires that the economic life of the structure be longer than the design service life when subjected to the design loads and environmental spectra. Compliance with this policy requires (a) understanding of the resistance of the structure to deterioration as a function of usage, environmental exposure, and in-service damage, and (b) establishment of limits of deterioration that will preclude functional impairment of the structure.

The residual strength and durability requirements are currently being met on production composite structures, such as the F-15 speed brake and F-16 horizontal stabilizers, by a proof-test methodology based on the wearout model. In this approach, the composite structure is nondestructively evaluated (NDE'd), proof test to a load level above the design load level, and re-NDE'd. If the post-proof-test NDE indicates no growth of damage as a result of the proof test, the structure is accepted for service. The structural integrity criteria are assumed to be satisfied. Since the proof test approach is based on the wearout model, it suffers from the weaknesses of the wearout model which tracks damage accumulation through changes in residual strength and not through the observation of damage accumulation sites within the structure. Any change in fatigue spectrum would require an extensive test program to characterize the new expected residual strength distribution.

The Air Force Flight Dynamics Laboratory has on-going in-house and contractual programs examining a systematic approach to the development of procedures for assuring the

strength and durability of composite structures. The approach consists of documenting (a) the damage accumulation process in the composite, and (b) the residual strength as a function of the fatigue load and environmental histories. The damage documentation is done through the use of NDE techniques developed by the Air Force Materials Laboratory (AFML). As part of its program to establish damage methods in composites, the AFFDL is working with AFML NDE personnel in the development of three-dimensional x-ray methods for damage inspection of advanced composite structures.

A current contractual program is examining the damage accumulation process and its effect on the residual strength. These results will be used as input into analysis procedures being generated in-house and under contract to predict the residual strength as a function of the accumulated damage. If the preliminary efforts turn out to be reasonable, the approach of tracking damage accumulation will be pursued in a systematic manner. The approach outlined above is expected to form the foundation of a methodology for assuring the strength, damage tolerance, and durability of advanced composite structure for primary airframe applications.

Superplastic Forming/Diffusion Bonding Titanium Design Concepts

Titanium is an excellent material for aircraft application because it is strong, lightweight, and corrosion resistant, and it maintains good structural properties at high temperatures. In the past, the use of titanium in aircraft structures has been limited because it is a relatively expensive material and it has been difficult and costly to fabricate. This situation has, to a large extent, resulted from attempts to adapt basic aluminum design and fabrication techniques to titanium structures rather than utilizing the unique properties of titanium to advantage in both design and fabrication. The Built-Up Low-Cost Advanced Titanium Structures (BLATS) programs currently on-going in AFFDL will attack this problem by applying several of the most recently developed titanium technologies (such as superplastic forming/diffusion bonding, isothermal forging, and powder metallurgy) to major airframe structure with a goal of reducing the acquisition cost of conventional titanium structure by 50% and reducing the weight by 30%.

The major thrust of the program is toward the application of superplastic forming with concurrent diffusion bonding (SPF/DB). The SPF/DB process combines two natural phenomena of titanium which occur in the same temperature range (1650 to 1750 degrees Fahrenheit) into a one-step operation. At those temperatures, titanium becomes very plastic or pliable (superplastic) and when pressure is applied, it can easily be formed into the desired shape. In addition, titanium parts can be bonded together without adhesives (diffusion bonding) at those temperatures by the diffusion of the titanium molecules under pressure. This results in a joint that is as strong as the surrounding material. This process permits the manufacturing of complex parts in a one-step operation without extensive machining, mechanical fasteners, or assembly operations. Additionally, many detail parts which are normally fabricated individually can be eliminated, thus, reducing the cost of the final product.

The high buckling efficiency of the SPF/DB skin and shell designs permit elimination of numerous elements, and in turn, the elimination of thousands of fasteners. Preliminary cost and weight evaluations of a superplastically formed, corrugation-stiffened keel web section designed for the F-15 aft fuselage indicate a significant reduction in detail parts (75 to 4) and fasteners (1420 to 71) and an acquisition cost reduction of 77%.

Active Flutter Suppression For Aircraft With Stores

High performance fighter-attack aircraft are required to carry many combinations of external wing-mounted stores to obtain operational effectiveness in the combat role. During the design and development phase of most of these aircraft, the types of external stores carried were limited to those stores and store combinations specified in the original design. Subsequent utilization of the aircraft, development of new tactics, and design of new stores (weapons) results in wing/store configurations which are flutter-critical within the previously established operational flight envelope. Additional speed placards, wich are established 15% below the flutter speed, are currently the only available means of preventing flutter under these new operational configurations. This technique generally results in significant losses in available performance. It is anticipated that advanced fighters, regardless of initial mission requirements, will

experience similar growth patterns. A decrease in the operational flight envelope due to wing/store flutter placards could significantly reduce survivability and mission effectiveness.

The AFFDL has a planned program to demonstrate the capability of an active feedback control system which is adaptive to flight conditions and aircraft store configurations for preventing flutter caused by the carriage of wing mounted weapons. The program will also provide this technology for application to contemporary and advanced USAF fighter aircraft.

The capability improvements are intended to expand the flight envelope of fighter aircraft with any store and/or store combinations, which will result in an increase in survivability and an improvement in operational effectiveness. This improvement is indicated by the shaded area of the curve in Figure 10. This includes any original design flutter placards as well as subsequently established operational limits. Analyses indicate that for conventional aircraft such as the F-4, the removal of the external store flutter placard could provide 33% improvement in speed. Based on this envelope expansion, survivability through intense and sophisticated Anti-Aircraft Artillery threat for a low-level pop-up and dive bombing maneuver, would be improved 25%. The potential for further speed expansion and increases in survivability is greater for advanced fighter aircraft with more efficient structural designs and improved capabilities.

Improved Crack Growth Prediction

Currently within the Air Force, airframe life predictions are based on a crack growth damage accumulation package which interrelates the following elements: (a) initial flaw distributions, (b) structural design effects, (c) flight history, (d) basic crack growth material properties and (e) failure of life-limiting criteria. The damage accumulation package must include a model capable of describing the first-order effects of each of the above elements.

There are three programs currently underway in the AFFDL to extend the current capabilities of the Air Force to provide more accurate life predictions for airframe structures. The first of these programs, durability methods development, seeks to assess the effects of

distributions of initial flaws such as material imperfections and manufacturing defects on the subsequent life of airframe structures. Particular attention is being paid to structural effects such as fasteners and flaws at holes. A second program, force management methods, is primarily concerned with aircraft usage or flight history effects and the determination of required parameters to be recorded for improved individual aircraft tracking programs. The third program, improved methods for predicting spectrum loading effects, has as its goal the development of an overall USAF capability for predicting the influences of each of the factors mentioned above. To ensure the most cost effective airframe structures possible, this capability will be applicable from the earliest stages of preliminary design, through the detailed design stage and into the individual aircraft tracking and service life extension programs.

VEHICLE EQUIPMENT

The objective of the Laboratory effort in vehicle equipment is to provide demonstrated technology advances in conventional and alternate flight vehicle takeoff and landing, windshields and transparency enclosures, emergency crew escape, environmental control, combined environment reliability testing, and protection against hostile, man-made, and natural environments. The Laboratory provides a strong technological base in the vehicle equipment technologies for design and performance assessment and for supporting other organizations of the Flight Dynamics Community. Specific goals also include reducing life cycle cost and fuel requirements of Air Force flight vehicles, increasing the safety of flight, reducing the vulnerability of flight vehicles in a hostile environment, and improving operational capabilities and self-sufficiency.

Recent accomplishments, technical gaps, and future objectives in Vehicle Equipment technology are shown in Figure 11. These programs are briefly described below.

Advanced Environmental Control System

The Advanced Environmental Control System (AECS), shown in Figure 12, was developed to provide an improved avionics environment with no increase in aircraft penalties related to environmental control systems. The AECS used

off-the-shelf components, current technology applied to new components, and advanced development components. Innovations include extensive use of fuel as a heat sink, moisture removal with an advanced rotating high pressure water separator, use of an advanced variable geometry air cycle machine, and a regenerative process for more efficient refrigeration. The AECS was designed to fit and was tested on a McDonnell Aircraft Company F-15 test bed aircraft. The AECS accumulated 78 hours of operation in the F-15 in hot and humid environments. A significant potential for aircraft life cycle cost saving was demonstrated. Cost savings in fuel were demonstrated as a result of lower ram and bleed air requirements. The benign environment produced by AECS demonstrated a capability to reduce avionics overhaul and maintenance costs. The tests were successful, showing that AECS can deliver cool, dry air to the avionics throughout the F-15 operating envelope with decreased penalty to the aircraft.

Lightweight Impact Resistant Windshield, F-111

The AFFDL began development of a Bird Impact Resistant Transparency (BIRT) system in March 1973. In slightly more than three years, the concept was developed, tested, flown, and put into production for the F-111E aircraft.

In response to the need to reduce the weight penalty for bird impact resistance, the AFFDL began development of an Alternate Design Bird Impact Resistant Transparency (ADBIRT) system in July 1976. Within 18 months, the ADBIRT design concept was developed and flown in an Initial Operational Test and Evaluation (IOT&E). This meant the reattainment of the entire F/FB-111 aircraft mission envelope and maximum operational readiness and effectiveness without a costly weight penalty. This also provided technical stepping stones and guidance toward meeting the total technology needs in this area involving design, materials, repair and maintenance, durability, and cost of ownership. The ADBIRT system demonstrated the capability to withstand the impact of a four-pound bird at speeds of at least 500 knots regardless of the impact location. In contrast, the BIRT's capability at the aft edge (most adverse impact location) is 425 knots against the impact of a four-pound bird. The existing glass windshield capability is less than 250 knots. The ADBIRT system will be incorporated into the F-111 fleet retrofit program and will replace the BIRT system on an attrition basis on the F-111E and F aircraft. The technology demonstrated on the F-111 aircraft has shown that high structural performance

can be attained from low molecular weight transparent materials without severe weight penalties or sacrificing other system requirements. The AFFDL development of an ADBIRT system for the F/FB-111 aircraft resulted in: (1) improved bird impact resistance over the current production(BIRT) system; (2) a 40-pound reduction in the weight penalty for bird resistance, which negated the need for an $80 million recertification of the F/FB-111 aircraft ejectable crew module; and (3) a production cost reduction of approximately 25%.

Cryogenic Cooler For AIM-9 Missile

Sidewinder (AIM-9) missiles employ high pressure gas discharge, open cycle, cooling of the infrared sensitive detector. The open cycle cooling system has a finite cooling capacity and results in expensive ground servicing due to high pressure gas storage limitations.

The AFFDL developed a closed cycle cryogenic cooler which provides a continuous operationally ready capability. The mechanical compressor of the cooler is driven by 400 Hz aircraft power and develops a sinusoidal pressure pulse which is transmitted to the refrigerator expander component by a thin gas transfer line. During each pressure pulse, the expander removes a small increment of heat from the gas. This process, repeated many times each second, quickly cools the remote end of the expander to minus 320°F.

The closed-cycle refrigerator completed extensive environmental and flight testing (F-4, F-5, F-16) and is currently undergoing tests to verify reliability and life of the unit. It will eliminate the requirement for expensive ground servicing of the stored high pressure gas system and is expected to provide 750 hours of maintenance-free operation. The success of this effort is attested to by the designation of the improved Sidewinder missile with closed-cycle refrigerator as the AIM-9M, replacing the earlier AIM-9L models.

Survivability/Vulnerability Design Criteria

Combat survivability considerations have long been neglected in the design of USAF weapon systems, and many bitter lessons have been learned at the cost of materiel and lives. Many of the efforts to rectify these survivability deficiencies in our present systems are costly,

not only in funds, but also in system performance and effectiveness. For example, to fit one survivability modification kit on the F-105 costs about $63 thousand and adds 809 pounds to the aircraft.

Survivability, a military design discipline, requires specific design criteria. These criteria should be based on the system's mission and the anticipated threat environment, and trades should be made to provide a well balanced, combat-effective, survivable weapon system.

Nonnuclear survivability design criteria result in an intrinsically hardened weapon system, reducing the need for added protection such as armor. This reduction in added protection results in trading off protection weight for greater payload, range, and maneuverability. By integrating survivability into the weapon system design, fewer aircraft will be lost, thus reducing the total replacement costs. Combining the high survivability rate with greater payload and range has the effect of increasing the aircraft's capability to kill more targets per sortie. Good survivability design criteria will result in a combat effective, survivable weapon system.

Viable Replacement For Conventional Landing Gear

The Ground Effect Takeoff and Landing System (GETOL) concept was developed in the early 1950s as augmentation for VTOL/VSTOL aircraft for runway operations. It developed into a general takeoff and landing system for conventional aircraft in the early 1960s in order to expand aerial operations to amphibious and forward operating base environments. Early development ended when specific mission requirements for a GETOL aircraft were not identified. At the time development was terminated, GETOL aerodynamic theory had not been developed and cushion-borne stability was not completely understood. Had these problems been solved, GETOL could now be a viable alternative to the Alternate Aircraft Take-Off System (AATS), aircraft dispersal, and STOL tactical airlift. Off-runway and battle-damaged runway operations have been identified as a required operational capability. GETOL is an attractive alternative to provide that capability.

Atmospheric Electricity Hazards

Advanced flight vehicles will carry digital microelectronics subsystems to increase the effectiveness of flight-/and mission-essential avionics. In addition, increasing amounts of non-metallic structural materials will be used to reduce flight vehicle weight. Microelectronics are highly susceptible to disturbance or damage caused by lightning strikes and static discharges. Non-metallic materials offer less shielding than metals against electromagentic disturbances caused by atmospheric electricity.

In order to establish a viable technology base to adequately assess the susceptibility or vulnerability of flight vehicles to the hazards of atmospheric electricity, three areas can be identified in which additional research and development work needs to be accomplished: (1) the definition and characterization of the threat, (2) the analytical modeling of threat/flight vehicle coupling, and (3) the development of realistic laboratory testing techniques for the verification of the effectiveness of hardening or protection schemes.

Near-term objectives are: (1) to acquire in-flight data to better understand, define, and characterize the lightning/flight vehicle interaction environment, (2) to develop an improved lightning/flight vehicle interaction arc channel model which adequately reflects the time history and spatial content of the event, (3) to improve ground-based test methods to more adequately simulate the natural hazards environment and to reduce or eliminate spurious ground-based effects, and (4) to develop electromagnetic pulse interaction and coupling models and computer codes to predict lightning electromagnetic energy penetration and induced voltage and current pulse levels on internal cables and circuits.

Alternate Aircraft Takeoff System

To fully satisfy mission requirements, Air Force aircraft must have the capability to be rapidly launched from existing airfields, even after airfield pavement systems have been damaged by enemy bombs or other weapons. The advent of aircraft able to accurately deliver devastating weapons loads at supersonic speeds has increased the vulnerability of and the damage which can be inflicted on

airfield pavement systems. As a result, the ability to rapidly launch existing aircraft is dependent upon the development of new, rapid methods to repair airfield pavement systems and/or the development of an Alternate Aircraft Takeoff (launch) System (AATS).

Typical approaches for the runway damage threat have been to develop advanced runway repair schemes and kits. With advancements in runway weaponry, however, these techniques have approached the limits of their capabilities.

In the past, a significant amount of work has been done aimed at the development and improvement of alternate takeoff (launch) systems to satisfy other aircraft operational requirements and/or the needs of specific aircraft. The rapid launch of aircraft from Navy aircraft carriers is a highly developed technology. Some work has been done to investigate the use of fixed and mobile ground based systems to launch aircraft. RATO and JATO methods have been developed to shorten the takeoff distance of existing aircraft. Aircraft have been launched from wheel-equipped and rail type trolley systems. New air cushion systems and hovercraft technologies have been developed which offer unique potential ability for aircraft takeoff from any relatively level ground surface including pavement, grass, snow, ice, and even water.

Closed Cycle Environmental Control System

Current open-cycle environmental control systems are very inefficient because they continuously extract high energy engine bleed air, process it, cool equipment, then vent it overboard. At cruise conditions, an advanced fighter requires 285 HP to produce 35 HP of refrigeration. A Closed-Cycle Environmental Control System (CECS) can provide the same cooling with a 40 HP input. Therefore, a major thrust in AFFDL is to develop a CECS which will reduce environmental control system penalties and increase aircraft performance. CECS eliminates engine bleed air as its primary energy source and it continuously recirculates equipment cooling air. A CECS will reduce engine bleed air extraction and ram air drag penalties by 80% and, therefore, reduce fuel consumption. A CECS will extend cooling system capabilities to high speed aircraft and reduce bleed air specific fuel consumption penalties for advanced engine design. It will also simplify crew member nuclear protection/survivability by significantly reducing the amount of air to be filtered/decontaminated.

Integrated Thermal/Avionics Design Capability

The Integrated Thermal Avionics Design/Computer Aided Design (ITAD/CAD) System is a responsive interactive computer aided design and analysis system. It is based on existing computer programs, efficiently organized by an executive program and integrated with a data management system. It uses weapon system life cycle cost minimization criteria to perform thermal management of avionics modules and integration of avionics with airframe environmental control systems. An important element of the CAD System is an Optemp computer program, which positions electronic components on air cooled circuit boards to maximize reliability based on thermal considerations.

Optimized thermal management of avionics modules will become possible by extending the AFFDL-developed Optemp program to handle the most important heat removal approaches. This program also allows consideration of environmental stresses in part-placement decisions, as well as interactions with electrical interconnections.

Optimized airframe integration will be accomplished by a capability to design aircraft environmental control systems to provide maximum cooling consistent with avionics reliability and weapon system life cycle cost minimization goals while preserving performance.

FLIGHT CONTROL

The objective in Flight Control is to develop flight control technology and design methods which add new dimensions to vehicle control capability and physically match the pilot with the performance and dynamic characteristics of the vehicle, its armament, and the mission systems. In addition to providing support to other organizations, a strong technical base development is maintained in flight control design techniques, specifications, and criteria. Specific goals include: developing flight control/man-machine interaction technology considering total mission environment; enhancing mission effectiveness through the development of innovative and cost effective technologies for safely expanding the operational capabilities of flight vehicles; developing technology to reduce acquisition and life cycle costs and to improve vehicle survivability; and maintaining analysis and simulator

capabilities and experimental testing methods for flight vehicle systems evaluations.

The Laboratory's recent accomplishments in the Flight Control area are listed in Figure 13, along with technical gaps and future objectives. Each program is discussed below.

Completion of Control Configured Vehicle Fighter Flight Test

The Air Force Flight Dynamics Laboratory's Fighter Control Configured Vehicle (CCV) Advanced Development Program was conducted to develop and evaluate advanced control concepts for improving fighter aircraft mission effectiveness. By decoupling the translational and rotational degrees of freedom, a number of new control degrees of freedom were provided in an existing high-performance fighter. Control modes selected for implementation were identified from previous research efforts as possessing the potential for significantly improving fighter aircraft performance.

A modified YF-16 aircraft was used to flight-demonstrate the decoupled control modes. Higher levels of direct force control were achieved by the aircraft than had previously been flight tested. The direct force capabilities were used to implement seven manually controlled, unconventional modes on the aircraft, allowing flat turns, decoupled normal acceleration control, independent longitudinal and lateral translations, uncoupled elevation and azimuth aiming, and blended direct lift. Use of these unconventional control modes not only provides the pilot with unique aircraft maneuvering capability, but implicitly provides gust alleviation, improved tracking, and cross-wind combat capability. This program provided the first true test of the utility of these new capabilities.

The unconventional control modes were flight-evaluated during simulated operational tasks, such as air-to-ground bombing and strafing, and air-to-air tracking and defensive maneuvering. Flight testing identified many actual and potential applications for these control modes; the results of this effort provided the foundation for new concepts of vehicle control, and for the design and development of task-tailored multimode CCV control modes for future fighter aircraft.

Integrated Flight/Fire Control Systems

Integrated flight/fire control (IFFC) concepts provide a means for dynamically coupling the director fire control system with the aircraft flight control system to automatically bring the weapon on the target. A joint AFFDL/Air Force Avionics Laboratory (AFAL) program (FIREFLY II) recently developed the system configurations for full or partial automatic control of a fighter aircraft in air-to-air (AA) and air-to-ground (AG) gunnery and bombing. These configurations were simulated in both piloted and nonreal-time digital simulations. Emphasis was placed on the terminal phase of weapon delivery where precision control is of critical importance.

A director fire control based upon a Kalman target state estimator is the basis for all configurations. The Kalman estimator was developed, for the most part, from previous AFAL programs. Establishing tracking accuracy requirements for the director was the objective of the program, rather than designing to match specific sensor specifications.

The system accommodates a variety of levels of pilot participation in the control tasks, ranging from complete manual, to pilot-aided semiautomatic control, to full automatic control. Division of control authority between the pilot and the automatic system can allow safe control in air-to-air gunnery where very high load factors are necessary. Effective, limited authority operation in other weapon delivery tasks is also a system capability. Additional modes of operation include automatic control of certain channels (e.g., yaw rate) while the pilot controls the others, and pilot sharing of precision gun aim. Emphasis in the piloted simulation, however, was on full manual and full automatic control, with only limited data taken on the additional mode of operation.

A nonlinear control and maneuvering concept has been developed for air-to-air gunnery to maintain effective weapon line pointing during large evasive maneuvers and precision control during high-angle-off and high-closing-rate attacks. For air-to-ground gunnery and bombing, significant enhancement of survivability and weapon delivery accuracy was achieved.

The Air Force Flight Test Center and the Tactical Air Command spent two weeks in evaluating the system in an F-15 and F-16 simulation. Data from 800 air-to-air passes, 300 air-to-ground gunnery passes, and 260 bombing passes were recorded from subsequent analysis. Principal conclusions from the simulation tests were:

(1) Integrated flight/fire control offers a major improvement over current air-to-air gunnery systems or over a manual director system based on the same target state estimation;
(2) Exchange ration (kills/losses) in a high threat environment can be increased by an order of magnitude when the FIREFLY system is used to its full potential with current high-velocity 300mm gun systems; and
(3) The exchange ratio for bombing in a high-threat environment can be significantly improved.

This effort is the direct predecessor of the AFFDL/AFAL IFFC/FIREFLY Advanced Development Program. Also, both Laboratories have begun exploratory development efforts at integrating advanced fire control systems, missiles, and smart weapons with conventional and advanced flight control systems.

Aircrew Workload

In support of the Air Force's Advanced Medium STOL Transport (AMST) Program, the Air Force Flight Dynamics Lab conducted an effort directed at defining the crew system design criteria applicable to the new aircraft. The work, entitled Total Aircrew Workload Study (TAWS), was a full-mission simulation effort in which experienced C-130 pilots from Military Airlift Command operational squadrons flew simulated tactical resupply sorties. Based on the assessments made by the pilots, the TAWS program concluded that if a two-pilot crew is to successfully complete the AMST mission, a highly integrated navigation management system, an integrated communications system, a formation flying system, and visual guidance augmentation are all required.

A significant outcome of the effort was the methodology and criteria for the design of crew stations with two-man crews. Recent emphasis was placed upon this concept for application to the KC-135 aircraft modification.

Software Technology

To enhance the effectiveness and safety of future digital control systems, several critical software technology areas have been identified for further development. The safety aspect is important since future aircraft with active control systems may not be able to maintain safe flight if the flight control system fails. Increased standardization of digital systems through the development of techniques of programming for functionally structured systems simplifies the tedious software verification process. Higher-order languages must be investigated to improve efficiency and reliability and to investigate the impact these languages have on system development costs and safety. Software verification and validation methods must be developed which are automated and that provide complete coverage to eliminate the possibility of latent failures due to undiscovered problems.

Controllability/Observability Theories

With the advent of digital processors, the long-known concepts of controllability and observability can now be used to obtain more information about the state of a flight vehicle from the same number of sensors (or the same amount of information from fewer sensors). This implies tht the same degree of reliability can be obtained with as much as 30% reduction in sensor hardware by replacing some of the redundant components with computational algorithms. The fundamental principle that allows this type of replacement is the fact that many sensors provide, in addition to their intended measurements, additional "coupled" information on other states which, when processed correctly, can be used to reconstruct the primary output of some other sensor. Similarly, if one control surface input is lost, it may still be possible to control all of the vehicle state variables through computed combinations of other control surface deflections.

Future control concepts that integrate the pilot with the flight vehicle will benefit from the use of Controllability/Observability Theories. Reuse of these theories to analyze chaotic conditions can indicate which information systems are reliable, which are inadequate, and which mission objectives are no longer obtainable. Such analysis may lead to improved system capability under these conditions through implementation of appropriate control system capability.

Functional Integration Architecture Methodology

Architecture in present day systems is determined primarily by equipment requirements and configurations. With the advent of six-degree-of-freedom flight, sensor blending, and computational capability, the flying mechanism characteristics can be dynamically optimized to mission-specific task needs. This allows improved use of existing equipment by functional integration. The methodology for this approach to dynamic optimization must be developed.

Integrated Control

Distinct cost reduction and performance benefits can be achieved through integrated control concepts. These concepts provide the process and methodology for functional and dynamic integration of crew, sensors, vehicle, and weapons. Weapon systems with integrated control will be capable of continued operation in a chaotic environment even though information through C^3 channels is temporarily disrupted. This weapon system will provide more time to the pilot for making mission-essential decisions through relieving him of the time consuming tasks of reading, storing and correlating data from several displays prior to making the decision. Inherent in such a system is the ability to optimize weapon system response characteristics to the unique needs of different mission segments, thereby providing the best possible response characteristics for the required tasks in that segment. The intrinsic capabilities of the integrated control concept are as follows:

Improved performance capability.

Automated design process.

Focused attention on information use.

Enhanced analytical redundancy.

Decreased vulnerability of flight control system to battle damage.

Minimized sensor complement.

Improved ability for examining of unconventional sensors and controllers.

Reduced sensitivity to sensor placement.

Distributed Microprocessors

There are many attributes of a distributed microprocessor system that make it an attractive candidate for flight control applications. A distributed system has fast response time, high throughput and high reliability. It can be easily made to be expandable, self-adapting to work load changes during flight, and real-time reconfigurable (self-healing). Distributed architectures are being analyzed and configured for best performance and cost effectiveness and a system will be fabricated using state-of-the-art microprocessors to demonstrate the concept. This effort will entail a simultaneous examination of the control laws, programming, numerical relationships, and life cycle ownership factors that form the basis for such a system.

State-Space Control Technology

Modern computer capability allows the use of state-space control technology that will permit efficient analysis, synthesis and development of a design method to maximize combat performance in advanced aerospace weapons sytems. State-space technology permits the combining of differential gaming theory with the control laws to achieve multi-dimensional flight path control for improved air-to-air combat positioning and air-to-ground munition delivery accuracy. This technology will permit exploitation of six degrees-of-freedom control capability to achieve weapon systems positions during complex maneuvers that are presently unachievable. Developing this technology will provide a tool for describing and comparing system architectures so that a desired capability can be achieved.

Total Mission Simulation

The AFFDL is applying engineering simulation as a cost-effective design tool for the formulation of advanced weapon systems concepts, the development of innovative subsystems, and the establishment of data bases, military specifications, and criteria. Simulators can accurately predict how advanced aircraft and their crews will perform throughout their flight envelope. Engineering simulation can realistically involve the aircrew in the design-decision process under a total mission environment. Sensitivity of total mission performance to natural and

battle-induced environments is heavily influenced by the aircrew's ability to interpret information received and to make real-time, accurate decisions. Technology alternatives for increased effectiveness, survivability, and safety can be accurately assessed under widely varying, but repeatable, combat situations with engineering simulators. Changes to aerodynamic agility, crew sizing, weapon complements, tactics, etc., can be realistically evaluated by engineering simulation methods prior to commitment to full-scale development and flight testing.

The AFFDL is expanding its Engineering Flight Simulation Facility to better accomplish total mission simulation. Libraries of mission tasks and advanced aircraft models will be programmed and available for rapid response evaluation with new technologies. Elements included in the total mission scenarios are shown in Figure 14.

AEROMECHANICS

The general objectives of the Aeromechanics efforts within the AFFDL are to advance technology in aerodynamics, aerothermodynamics, performance analyses, aerodynamic ground simulation, configuration research, aerodynamic/propulsion integration, and technology demonstration related to future manned and unmanned weapon systems. The Laboratory maintains a competent and competitive technology base and audit capability and provides support to other organizations in Aeromechanics Technology. Additional specific goals include: (1) Development of innovative technologies for aerodynamically tailored aircraft which include performance methodologies for increasing survivability, effectiveness and reduced costs; (2) Identification and development of technology drivers and performance methods for aerodynamic configured unmanned vehicles including missiles; (3) Development of propulsive/aeromechanic and weapons/airframe/aerodynamic integration for increased stealth, improved performance, and reduced costs; (4) Formulation of advanced flight vehicle concepts through missions effectiveness, cost analysis, system evaluation, and technology assessment; (5) Pursuit of technology demonstrators for high pay-off technologies; and (6) Maintaining a comprehensive aerodynamic ground test simulation capability.

Aeromechanics programs and concerns are listed in Figure 15 in terms of recent accomplishments, technical gaps, and future objectives. These programs are briefly discussed below.

Transonic Aircraft Technology

The Transonic Aircraft Technology (TACT) program has been successfully completed and the major results have been transmitted to the aerodynamic community by means of the Symposium on Transonic Aircraft Technology. The objective of the flight demonstration was to verify the aerodynamic benefits of the supercritical wing as predicted by wind tunnel tests on the 1/24-scale TACT model, and as illustrated in Figure 16. The demonstration was intended to increase confidence in supercritical airfoil technology and to encourage its use on new aircraft designs. The design objective of the TACT supercritical wing was to improve the transonic maneuvering capability without degrading cruise or supersonic performance. The flight demonstration was successfully completed with close agreement between the flight-derived aerodynamic improvements and those predicted from development wind tunnel data.

A second objective of the flight program was to establish procedures for correlating wind tunnel and flight test results. Correlation is defined as the degree of conformity between flight test and wind tunnel derived aerodynamic parameters. The objective of the correlation was to assess the wind tunnel test methods and adjustment procedures used to predict "estimated, full-scale aerodynamics". Attendance to the details of these procedures has resulted in good correlation over a large Mach number/dynamic pressure range for the TACT aircraft. The correlation of aerodynamic parameters included not only lift and drag, but also wing static pressures, buffet, and stability and control derivatives. The highly instrumented TACT aircraft has provided a test platform for various aerodynamic investigations with comparison to theoretical and model test data.

Propulsive Lift Enhancement

Recent propulsive-lift experimental investigations of advanced fighters indicate significant potential improvements in high lift and maneuver performance as a result of aerodynamic/propulsion integration. Various design concepts

are presently being evaluated by the Air Force, Navy, NASA, and industry. One concept, Vectored Engine-Over-Wing (VEO-Wing), a joint development of the AFFDL, NASA and General Dynamics, is a near-term application which vectors the full engine exhaust, from current technology engines, to achieve both supercirculation lift and leading-edge vortex augmentation.

The VEO-Wing design concept features a canard/wing arrangement with twin engines mounted over the wing. The nonaxisymmetric, vectorable nozzles are integrated with the trailing edge flap. Exhaust ports, located on the engine nacelles at approximately thirty percent wing chord, produce a spanwise flow for leading-edge vortex augmentation. With this geometry, lift augmentation is obtained from both the thrust deflection and spanwise blowing.

Powered model wind tunnel test results have validated that the VEO-Wing concept is capable of generating low-speed trimmed lift coefficients greater than four, and subsonic and transonic maneuver capability superior to that of a conventional design with equal wing loading and thrust loading. The high lift coefficients resulting from propulsive lift enhancement provide excellent short take-off and landing (STOL) performance (less than 800 feet). This STOL performance is obtained without penalizing the transonic maneuver characteristics or the supersonic capability.

Flow Field Computation for Practical Three-Dimensional Components

Computational Aerodynamics has progressed rapidly in recent years and has become a powerful tool for aerodynamics research and engineering development. AFFDL has successfully developed numerical methods to obtain solutions of three-dimensional, time-dependent Navier-Stokes equations for simple shapes. See Figure 17.

Using these methods, aircraft components such as nosetips, airfoil sections, delta wings, and wing-fin junctions, have been analyzed for realistic Mach numbers and Reynolds numbers. Improvements in the program are under development to include more efficient algorithms on today's super computers. Recent work on the CRAY-1 computer showed a speedup in the program of 128 times

over the CDC 6600. Calculation of a complete configuration will be achievable in the 1980s. Potentially, Computational Fluid Dynamics can revolutionize the design process and reduce the requirement for testing, shorten the design cycle, and decrease the development cost of flight vehicles.

Forward-Swept Wing Aerodynamics

An effort is underway both at AFFDL and NASA Langley Research Center to develop a data base for forward-swept wing aerodynamics. Approximately thirty years of aerodynamic development of aft-swept wings has taken place if allowance is made for a break in the effort during the sixties. Technology options have emerged such as supercritical wings, variable camber, aeroelastic tailoring, advanced transonic three-dimensional wingbody design, etc., all of which are keyed to developing aft-swept wings.

With the advent of aeroelastically tailored composite structure, the classical bend-twist coupling problem leading to structural divergence of a forward-swept wing can be overcome without substantial weight penalty. Furthermore, it is possible to optimize such a structure for a wide variety of configuration concepts.

Supercritical transonic wing technology provides a method to gain lift improvement by moving the shock wave further back. For a forward swept wing, this can result in a more highly swept shock for the same structural sweep. Recent AFFDL tests, conducted at the AEDC 4T facility, have verified the shock sweep effects, and several optimized configurations from minus 15 to minus 45 degrees of structural sweep will be tested in the near future.

Analysis of Unconventional Forces On Flight Vehicles

Many technologies in airframe-propulsion integration and flight control are emerging for highly agile tactical fighter/strike aircraft. These technologies, which include direct lift/direct side-force, fuselage pointing, drag modulation (or thrust reversal), thrust vectoring, etc., basically allow the six degrees of freedom to become decoupled, and allow new and unique flight modes to be considered for air combat or surface attack.

From exploratory and advanced development programs, the AFFDL is making progress on mechanizing these unconventional forces although there is still a long way to go.

The CCV-16 was an example of a flight test program which demonstrated the feasibility of direct lift/direct side-force concepts. Other unconventional modes are being considered in other AFFDL programs.

The ability to conduct analyses of these unconventional modes in the highly dynamic environment of air-to-air combat is deficient. Although we have a relatively good understanding of the aerodynamic and control characteristics of most of these concepts on a particular design, the ability to thoroughly analyze the total influence on effectiveness in air combat, particularly the multi-aircraft air engagement, is lacking. The days of the one-on-one dogfight where only two aircraft are engaged for long periods of time seems to be giving way to short, high intensity engagements of several combatants. This kind of engagement will place a higher premium on aircraft "quickness" or agility rather than the classic sustained maneuverability.

The AFFDL will initiate efforts to improve analysis of multiple-engagement air combat and try to validate the analysis with flight experience and the recent simulated trials conducted by the Department of Defense. These trials can be viewed as an experiment in multi-aircraft combat with specific initial conditions and constraints which can be modeled. After validation and correlation, analysis methodologies can be extended to unconventional aircraft control forces to better determine where new flight modes appear to have the highest payoff. This capability would reduce development and acquisition costs by technology screening and developing revolutionary tactics for future tactical fighters.

Effective STOL Aero/Propulsive Designs

Past attempts to develop fighters with a short takeoff and landing (STOL) capability have generally resulted in compromised designs. Various schemes have been proposed which have utilized lift engines, ejectors, lift/cruise engines, and rocket or jet assisted takeoff, in conjunction with increased wing area employing heavy, complicated high lift systems. Recent advances in non-axisymmetric nozzle technology provide the airplane designer with alternatives to the brute force approaches of the past. Integration of the aerodynamic surfaces with the propulsion system has been found to generate induced forces resulting in greatly enhanced aerodynamics. Through careful design, a STOL capability may be obtained without severe penalties in transonic maneuverability or supersonic performance.

Several aero/propulsive design concepts are currently being pursued under Government and industry sponsorship. The concepts include: vectored thrust/supercirculation, jet flap, upper surface blowing/externally blown flap, and vectored engine-over-wing. Each concept utilizes the propulsion system to augment the basic aerodynamics. Unique design challenges exist for each concept. Trim of the aerodynamic and propulsion-induced forces and moments becomes increasingly more complicated. Use of fly-by-wire flight/propulsion control systems is mandatory to provide artificial stability and control during STOL operation and meneuvering flight.

Aero/propulsive design technology is available for STOL application; however, there is a considerable gap in the experimental data base. Effective STOL fighter designs will result from careful integration of the aerodynamics with the propulsion systems. However, the final designs will require verification through wind tunnel and flight demonstration.

Increased Survivability Technology

Future tactical aircraft must be designed to reduce signature levels in order to be survivable in an increasingly hostile environment. See Figure 18. All observable elements are important, but for air-to-surface aircraft the radar and IR signatures are of most urgency. For air-to-air combat, the visual characteristics can be added.

The IR signature derives from the hot parts of the propulsion system, the exhaust plume, and airframe heating at high speeds. Benefits from nonaxisymmetric nozzles include shielding of hot parts and reduction of plume temperature through increased mixing with the free air stream. These effects are known and quantified for non-flight cases. A flight demonstration of nonaxisymmetric nozzles is required.

The radar signature of an aircraft varies with viewing aspect, and criticality depends on the threat. For example, the most critical aspect for a high speed penetrator is forward. The largest contributors are the engine air inlet cavities, the nose radar compartment, and the cockpit. Second to these are the nose shape, the lifting surface leading edges, weapons installations, and control surface gaps. Additional contributors such as skin discontinuties

(joints, access panels, etc.) and surface intersections are of little importance until the larger contributors are suppressed. Treatment technology is fairly well understood for inlets, radar compartments, and cockpits and depends primarily on availability of materials. The relationship between inlet placement and shape and low radar cross section from frontal aspects is not well enough understood to support vehicle preliminary design. The second-echelon contributors consisting of nose shape, leading edges, weapons installations, etc., are not well understood in a vehicle design sense. The interactions between radar return, aerodynamic impact, and vehicle shape have yet to be established.

The aft section contributors are similar to the nose, consisting of cavities, edges, etc., although treatment is complicated by the hot exhaust environment. Materials are not currently available for exhaust nozzle treatment.

The broadside returns are dominated by the vertical tail and fuselage shape. Geometry changes can be effective here, as illustrated in the figure. The aerodynamics of such designs are almost totally unknown in an experimental sense. Very little aerodynamic testing has been done.

In summary, designing for low radar signature will lead to planar surfaces with sharp, straight edges of minimum number; and engine air inlets will tend to be placed in poor locations from a performance point of view. Programs are required to quantify the limits to which the aircraft architecture can be dictated by radar cross section considerations while maintaining acceptable performance and handling qualities.

Aero Configured Missiles

The purpose of this work is to investigate the aerodynamic potential that can be obtained from incorporating innovative aerodynamic features into missile configurations. Missile improvements will be achieved by applying knowledge of lifting bodies and applying favorable interference to various aerodynamic shapes. Aerodynamics is related to performance goals through analysis and configuration classes which produce enhanced capabilities and which are analytically defined and validated through wind tunnel testing. Operational and physical constraints will be imposed on the configurations to assess their effects on the aerodynamic characteristics.

Air-launched missiles will benefit from the technology advancements through improved range factors for the longer range cruise type missiles and increased levels of maneuverability. Four configurations, a bench mark shape, a typical non-circular shape,and two favorable interference concepts, have been investigated. The favorable interference concepts had the higher L/Ds. However, configuration complexity increases launch weight.

Favorable Aerodynamic Interference Configurations

Aircraft capable of extended range while cruising at supersonic Mach numbers offer promise of a substantial improvement in military effectiveness, particularly for interdiction, strike, reconnaissance, and interceptor missions. The incorporation of favorable interference is an effective means to increase supersonic aerodynamic efficiencey. Considerable development of favorable interference technology has taken place in association with maneuverable, orbital entry, and hypersonic vehicle studies. The application of these favorable aerodynamic interference concepts to supersonic aircraft was selected for further exploration.

The Flight Dynamics Laboratory sponsored favorable interference efforts with the objective of identifying the various ways the concept could increase the aerodynamic efficiency of supersonic fighter type aircraft. The approach used aerodynamic analyses to evaluate the potential of the various concepts and identify the most promising. A parasol wing design was selected, using the flow field from the fuselage or propulsion units to create lifting pressures on the wing.

The principal aerodynamic parameters which benefit from favorable interference are the lift curve slope and, of course, the lift/drag ratio. A very encouraging result was that these improvements are evident over a wide range of Mach numbers. At the present time, an experimental program is being condicted at AEDC to measure the aerodynamic characteristics of a parasol wing configuration at supersonic and transonic speeds. Continued study of this concept for improving performance is planned.

Design Criteria For Low Radar Cross Sections

In order to survive in a hostile radar environment, future aircraft must be designed with significantly lower Radar Cross Sections (RCS) than contemporary air vehicles. Although much technology has been developed in recent years to reduce the RCS contribution of various airframe components, the findings have been applied, for the most part, after the weapons systems have been through the design and development cycle and often, after the systems have become operational. Generally, this type of reduction in RCS is more expensive, less efficient, and less effective than it would be had RCS been considered early in the design and development phase. To counteract this trend, an aerodynamic/RCS data base is needed. This information will provide an effective interface between RCS specialists and aircraft designers and will provide the tool necessary for configuring optimized military aircraft for which low radar signature is an important design parameter.

A three-phase program has been established in AFFDL to develop the needed aerodynamic and electromagnetic data required for configuring aircraft to reduce radar signatures. The program consists of an analytical section which uses available aerodynamic and RCS prediction methodology to screen the effect of aircraft architectural arrangement; an RCS test portion to acquire accurate signature information on a systematic variation of aircraft geometry features; and an aerodynamic test phase to quantify the performance decrements associated with designing airframes for a given RCS level.

The experimental information generated in the outlined program will be combined with existing aero/RCS data to establish a design handbook comprehensive enough to allow the aircraft designer to accurately shape aircraft with specified performance and RCS mission requirements.

Propulsive Lift/Integrated Exhaust Systems

Advanced aircraft configurations must rely on heavily integrated propulsion systems to provide optimum performance on demanding air-to-air and air-to-ground missions. Utilizing the propulsive jet not only as a thrust producer, but also as a source of direct and induced wing and body lift, these advanced configurations offer maneuverability and range far superior to comparable current aircraft.

To realize these improvements, a definitive research program has been initiated by the USAF to explore the the benefits of propulsive lift/integrated exhaust systems for advanced aircraft. Following configuration development, a series of wind tunnel tests exploring all aspects of integrated propulsion aerodynamics and total vehicle performance will be conducted to determine the configuration variables which impact the ability of the vehicle to meet prescribed mission performance. Using these data, the preliminary design engineer should be able to effectively integrate advanced propulsion systems which feature propulsive lift into advanced airframes to obtain optimum mission performance.

Technology Integration

At any given time one technical area may make a major contribution to the performance growth of a flight vehicle system. However, it is the integration of advances from several technologies which is essential to the deployment of the most effective weapon systems. The AFFDL has the responsibility for integrating and demonstrating Flight Dynamics technology within the Air Force. The major current effort accomplishing this function is the Advanced Fighter Technology Integration (AFTI) program described below.

The European battle scenario demands timely improvements in the lethality and survivability of USAF fighter aircraft. The AFTI-15/16 program will provide flight-demonstrated technology options for future fighter aircraft. This program will develop and flight-validate a set of advanced technologies (See Figure 19), to improve the weapon delivery accuracy and maneuvering survivability of fighter aircraft in both air-to-air and air-to-surface combat, using a modified F-15 or F-16 demonstrator. The advanced technologies include a digital flight control system, integrated flight and fire control systems, new unconventional control surfaces and ways to fly, and improved cockpit controls and displays. The common thrust of these technologies is to provide rapid and precise control of the weapon impact points while retaining the advantages of speed and maneuverability.

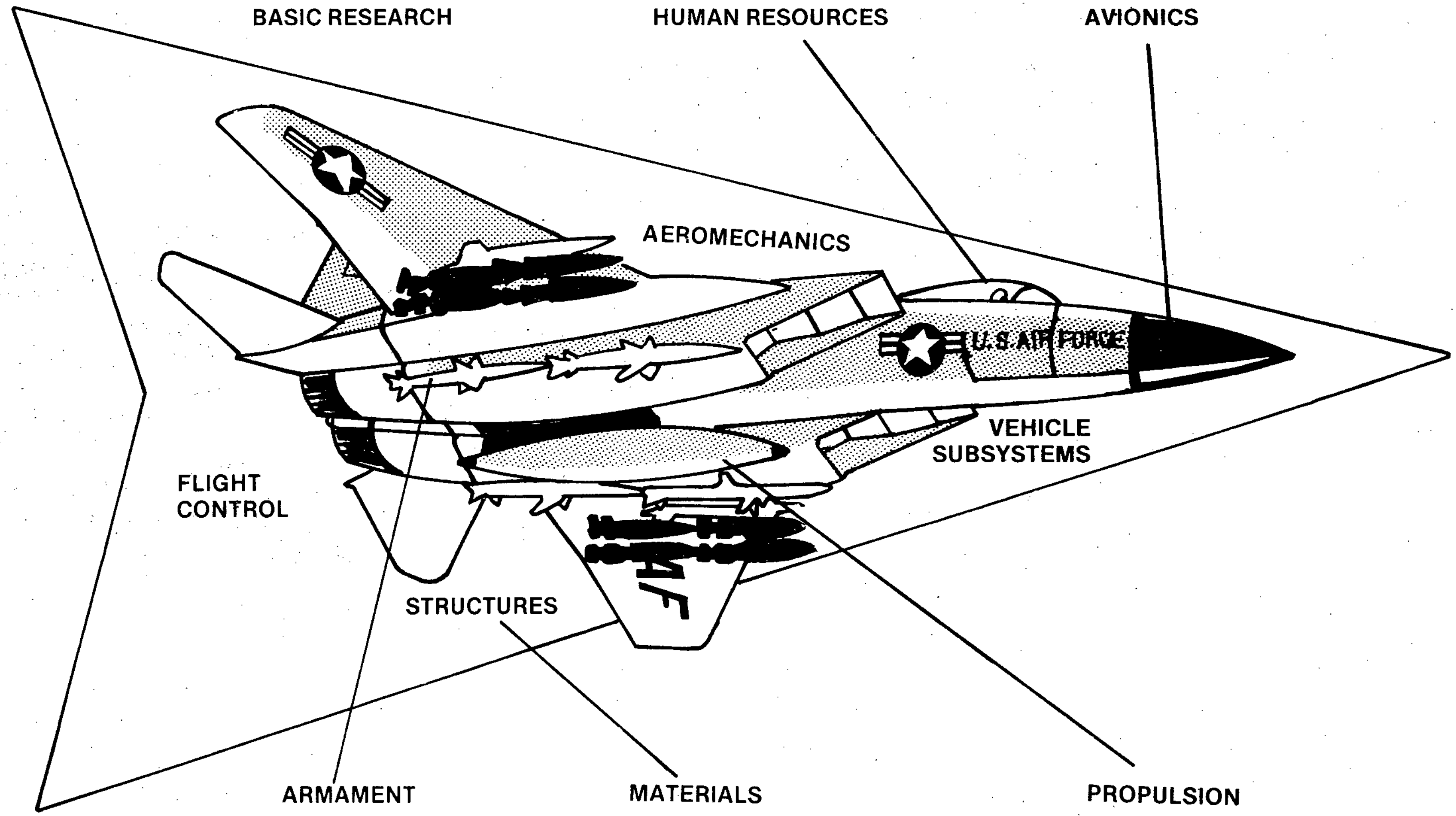
The digital flight control system will allow aircraft handling qualities to be tailored to the specific weapon delivery task (gunnery, bombing, etc.) and the surfaces will provide new ways to fly. The AFTI demonstrator can be

controlled to aim (rotate) the fuselage without changing flight path; and to directly translate up or down and right or left without the conventional roll-in and roll-out maneuvers. With new highly accurate target trackers and on-board target estimators, some or all of the weapon aim errors may be nulled rapidly, precisely, and automatically by coupling the flight and fire control systems. The pilot will have improved controls and displays to manage these new capabilities.

Expected improvements in air-to-air combat include a 4:1 increase in expected hits on enemy aircraft, a 3:1 reduction in time to first firing opportunity, and a significant increase in the number of weapon delivery opportunities. In air-to-surface combat, a 2:1 increase in delivery accuracy, together with a 10:1 increase in survivability against AAA defenses, is expected. The increase in survivability is due to the AFTI capability to deliver ordnance accurately on first target pass, while flying unpredictable evasive maneuvers not possible with present aircraft.

SUMMARY

This review of current activities and accomplishments in flight dynamics technology clearly indicates that progress in this field, initiated long before the first successful flight of the Wright Brothers, is continuing to provide dramatic advances. Each advance opens new avenues for exploration and provides more options for exploitation, so that the field is still expanding. The lists of outstanding accomplishments are impressive, and the lists of technical gaps and future objectives only hint at the potential for future work. The papers which follow this functional area review provide detailed examples of the variety and quality of the research which is continuing to increase our knowledge and capabilities in flight dynamics technology.


BASIC RESEARCH
HUMAN RESOURCES
AVIONICS
AEROMECHANICS
U.S. AIR FORCE
VEHICLE SUBSYSTEMS
FLIGHT CONTROL
AF
STRUCTURES
ARMAMENT
MATERIALS
PROPULSION

AIR FORCE

FLIGHT DYNAMICS LABORATORY
AERONAUTICAL SYSTEMS DIVISION
ARNOLD ENGINEERING DEVELOPMENT CENTER
FLIGHT TEST CENTER
FOREIGN TECHNOLOGY DIVISION
ARMAMENT DEVELOPMENT TEST CENTER

ARMY

ARMY AVIATION SYSTEMS COMMAND
ARMY AIR MOBILITY R&D LABORATORY

NAVY

NAVAL AIR DEVELOPMENT CENTER
NAVAL AIR SYSTEMS COMMAND
NAVAL AIR TEST CENTER
NAVAL SURFACE WEAPONS CENTER
NAVAL AIR ENGINEERING CENTER

AEROSPACE INDUSTRY
UNIVERSITIES
RESEARCH CENTERS

NASA

AMES RESEARCH CENTER
LANGLEY RESEARCH CENTER
DRYDEN RESEARCH CENTER
LEWIS RESEARCH CENTER

STRUCTURAL MECHANICS

- AIRCRAFT ENERGY EFFICIENCY PROGRAM (NASA)

FLIGHT CONTROL

- STRUCTURAL MODE STABILIZATION — YF-12 (NASA)
- SIMULATION TECHNOLOGY FOR ROTORCRAFT (ARMY)

AEROMECHANICS

- NON-AXISYMMETRIC NOZZLE/AIRFRAME INTEGRATION (NASA)

VEHICLE EQUIPMENT

- STANDARD MODULAR COOLER (ARMY)

STRUCTURAL MECHANICS

- METAL MATRIX COMPOSITES (ROCKWELL)
- ACTIVE FLUTTER SUPPRESSION (GENERAL DYNAMICS/LOCKHEED)

VEHICLE EQUIPMENT

- RESPONSE OF COMPOSITE STRUCTURES TO BALLISTIC IMPACT (BOEING)
- CYROGENIC RADIATOR (ROCKWELL)

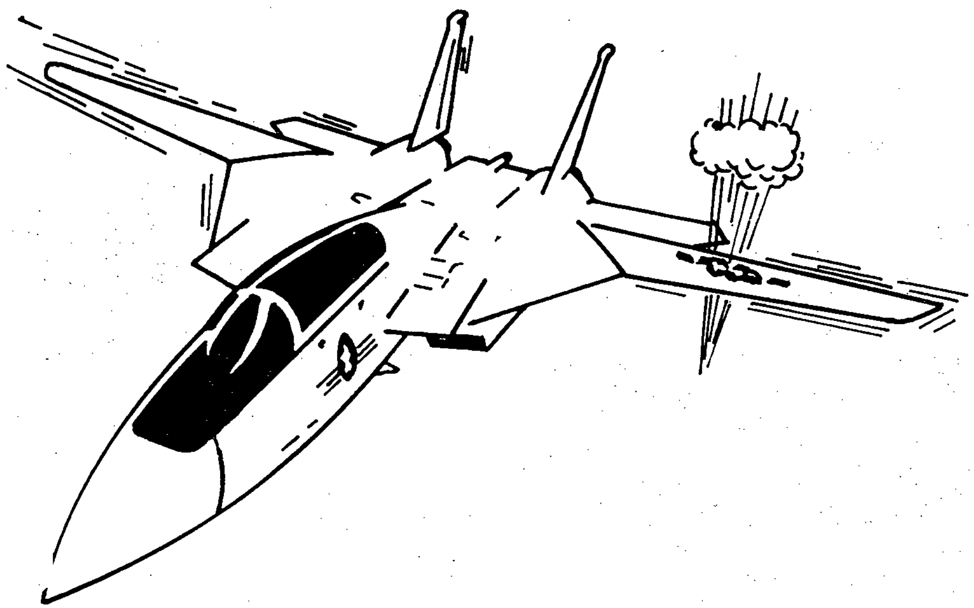
FLIGHT CONTROL

- REDUNDANCY MANAGEMENT (BOEING/LOCKHEED)

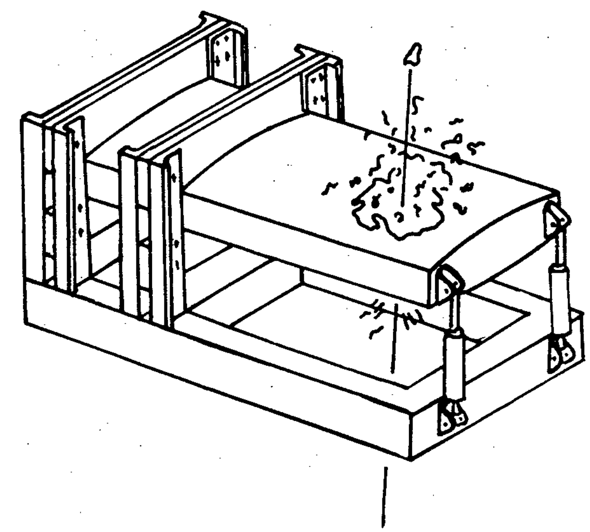
AEROMECHANICS

- VARIABLE GEOMETRY INLET FOR F-16 (GENERAL DYNAMICS)
- ADVANCED COMBAT AIRCRAFT CONCEPTS (BOEING)

THE COMBAT ENVIRONMENT



THE TEST ENVIRONMENT



OBJECTIVE: DEVELOP METHODOLOGY TO PREDICT EFFECT OF BALLISTIC IMPACT DAMAGE ON STRENGTH OF FIBER-COMPOSITE STRUCTURES

- GERMAN AEROSPACE RESEARCH ESTABLISHMENT (DFVLR, IABG) — FRG
- NATIONAL INSTITUTE FOR AEROSPACE STUDIES & RESEARCH (ONERA) — FRANCE
- ROYAL AERONAUTICAL ESTABLISHMENT (RAE) — UNITED KINGDOM
- VON KARMAN INSTITUTE FOR FLUID DYNAMICS (VKI) — BELGIUM
- NATIONAL AEROSPACE LABORATORY (NLR) — THE NETHERLANDS
- AERONAUTICAL RESEARCH INSTITUTE OF SWEDEN (FFA)
- EUROPEAN AEROSPACE INDUSTRY

STRUCTURAL MECHANICS

- ACTIVE FLUTTER SUPPRESSION (FRANCE, FRG, UK)

VEHICLE EQUIPMENT

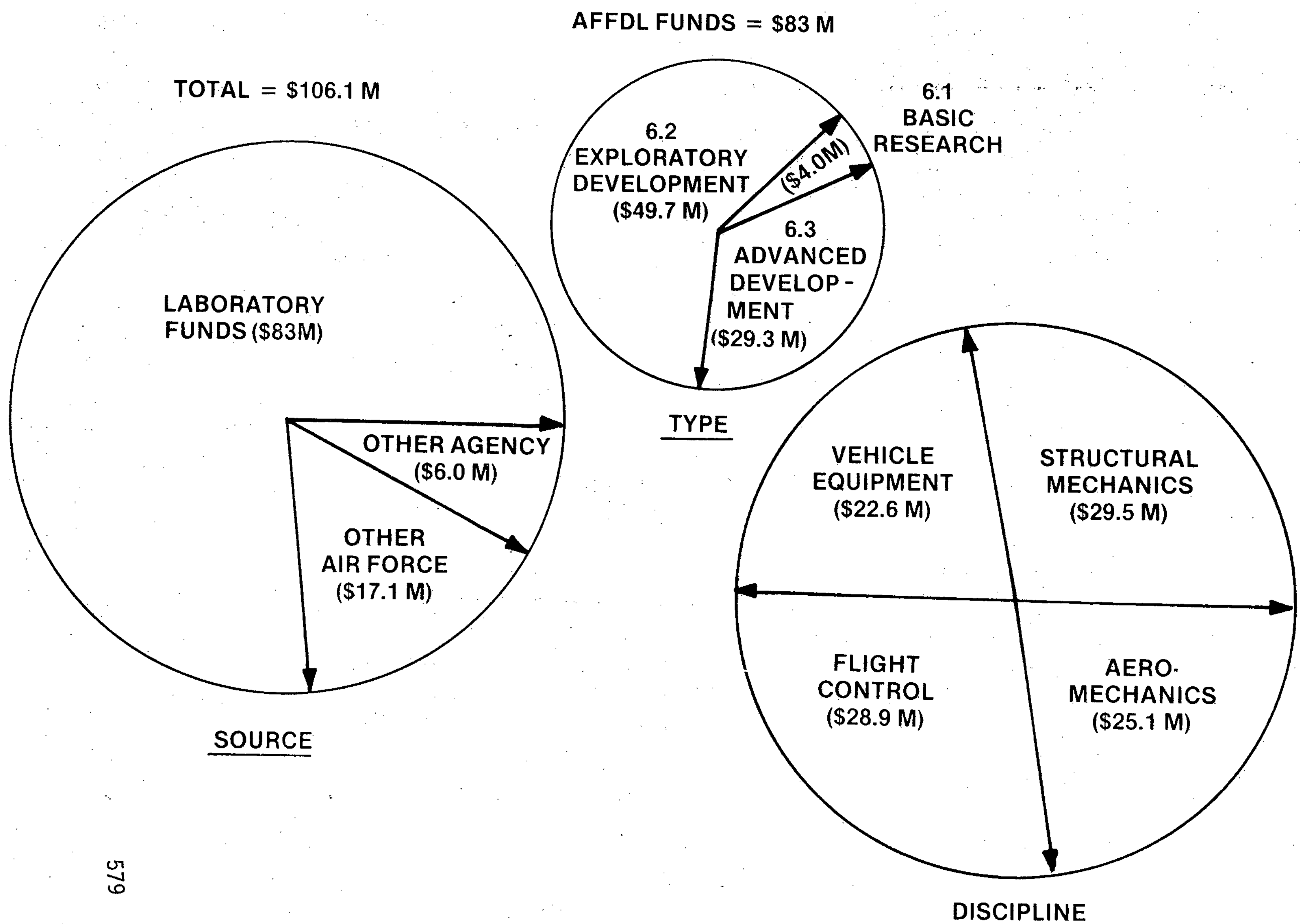
- POWDER PACKS FOR AIRCRAFT FIRE SUPPRESSION (UK)

FLIGHT CONTROL

- APPLICATION OF FLY-BY-WIRE AND CONTROL-CONFIGURED-VEHICLE TECHNOLOGIES TO MIRAGE 2000 (FRANCE), CONCORDE (FRANCE, UK) AND KFIR(ISRAEL)

AEROMECHANICS

- FUTURE GERMAN COMBAT FIGHTER (FRG)


TOTAL = $106.1 M
LABORATORY FUNDS ($83M)
OTHER AGENCY ($6.0 M)
OTHER AIR FORCE ($17.1 M)
SOURCE
AFFDL FUNDS = $83 M
6.1 BASIC RESEARCH
($4.0M)
6.2 EXPLORATORY DEVELOPMENT ($49.7 M)
6.3 ADVANCED DEVELOP - MENT ($29.3 M)
TYPE
VEHICLE EQUIPMENT ($22.6 M)
STRUCTURAL MECHANICS ($29.5 M)
FLIGHT CONTROL ($28.9 M)
AERO-MECHANICS ($25.1 M)
DISCIPLINE

RECENT ACCOMPLISHMENTS

- COMPLETION OF PABST DURABILITY VERIFICATION
- ADVANCED COMPOSITE HORIZONTAL TAIL, B-1
- EFFECTS OF FASTENER HOLE QUALITY ON STRUCTURAL LIFE

TECHNICAL GAPS AND FUTURE OBJECTIVES

- DESIGN/MANUFACTURING METHODS FOR ELEVATED TEMPERATURE SERVICE
- TRANSONIC AERO COMPUTATIONAL METHODS
- COMPUTER AIDED DESIGN TECHNIQUES
- STRENGTH AND DURABILITY REQUIREMENTS FOR ADVANCED COMPOSITE PRIMARY STRUCTURES
- SUPERPLASTIC FORMING/DIFFUSION BONDING TITANIUM DESIGN CONCEPTS FOR AIRCRAFT
- ACTIVE FLUTTER SUPRESSION FOR AIRCRAFT WITH STORES
- IMPROVED CRACK GROWTH PREDICTION

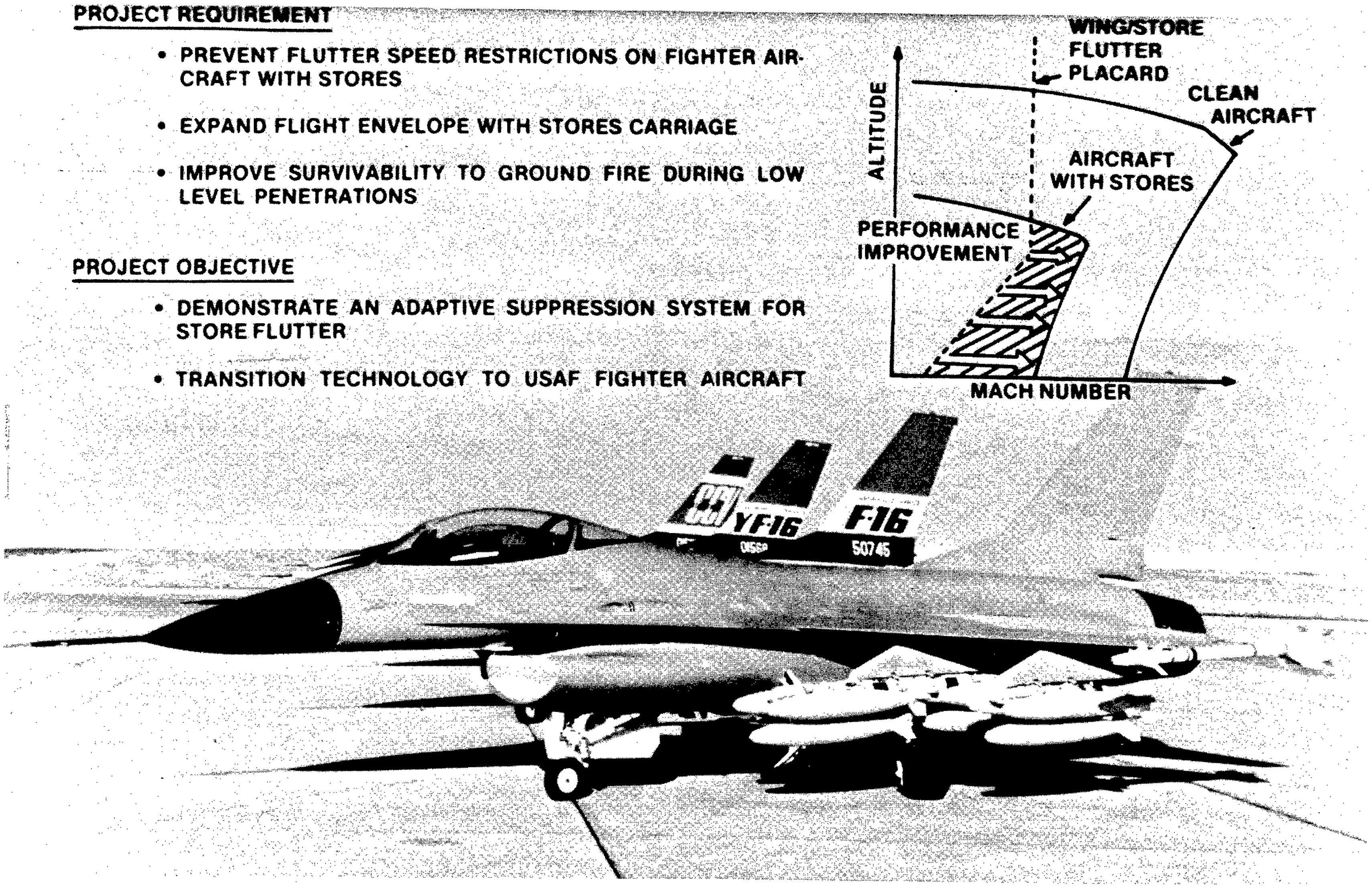


FIGURE 10. ACTIVE FLUTTER SUPPRESSION FOR AIRCRAFT WITH STORES

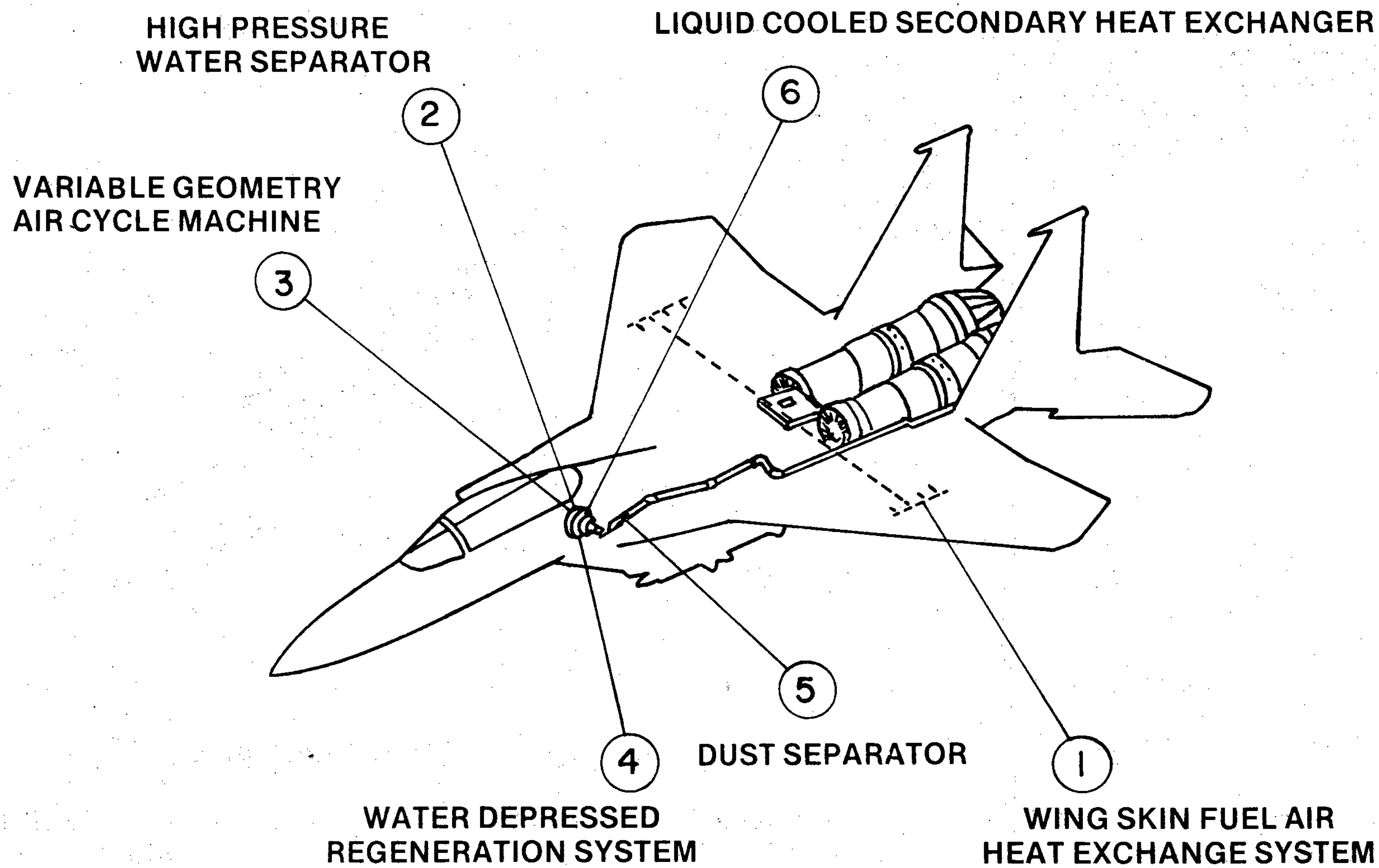
RECENT ACCOMPLISHMENTS

- ADVANCED ENVIRONMENTAL CONTROL SYSTEMS (AECS)
- LIGHTWEIGHT IMPACT RESISTANT WINDSHIELD (F-111)
- CRYOGENIC COOLER FOR AIM-9 MISSILE

TECHNICAL GAPS AND FUTURE OBJECTIVES

- SURVIVABILITY/VULNERABILITY DESIGN CRITERIA
- VIABLE REPLACEMENT FOR CONVENTIONAL LANDING GEAR
- AIRBORNE ELECTRONIC HAZARDS
- ALTERNATE TAKE-OFF SYSTEM
- CLOSED CYCLE ENVIRONMENTAL CONTROL SYSTEM
- INTEGRATED THERMAL/AVIONICS DESIGN CAPABILITY

FIGURE 11. VEHICLE EQUIPMENT PROGRAMS AND CONCERNS



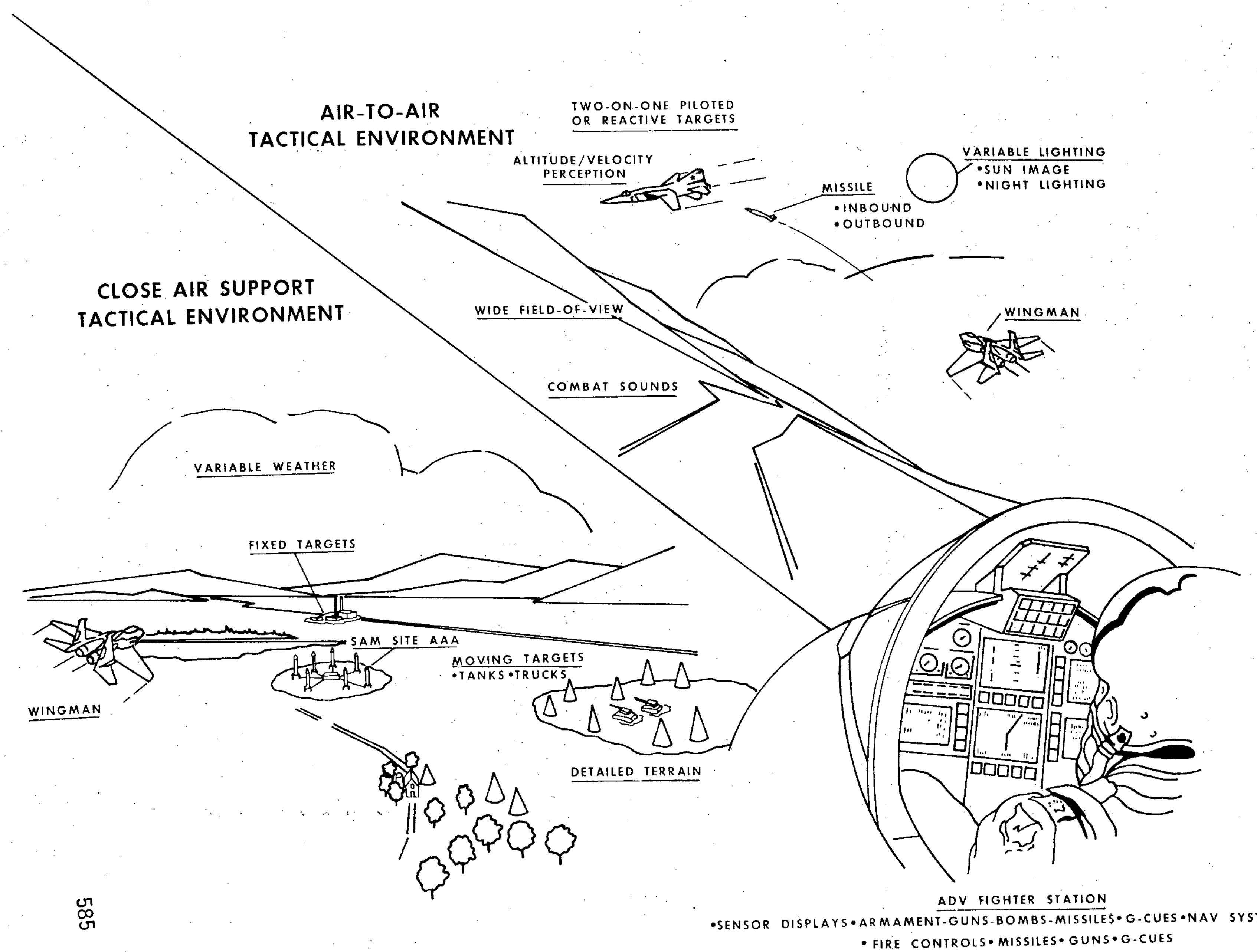
• MAXIMUM USE OF ON BOARD HEAT SINKS

RECENT ACCOMPLISHMENTS

- FIGHTER CCV FLIGHT TEST COMPLETE
- INTEGRATED FIRE/FLIGHT CONTROL
- AIRCREW WORKLOAD

TECHNICAL GAPS AND FUTURE OBJECTIVES

- SOFTWARE TECHNOLOGY
- CONTROLLABILITY/OBSERVABILITY THEORIES
- FUNCTIONAL INTEGRATION/ARCHITECTURE TECHNIQUES
- INTEGRATED CONTROL CONCEPT
- APPLICATION OF DISTRIBUTED MICROPROCESSORS
- DEVELOP STATE/SPACE CONTROL TECHNOLOGY
- TOTAL MISSION SIMULATION


AIR-TO-AIR
TACTICAL ENVIRONMENT
TWO-ON-ONE PILOTED
OR REACTIVE TARGETS
ALTITUDE/VELOCITY
PERCEPTION
MISSILE
•INBOUND
•OUTBOUND
VARIABLE LIGHTING
•SUN IMAGE
•NIGHT LIGHTING
CLOSE AIR SUPPORT
TACTICAL ENVIRONMENT
WIDE FIELD-OF-VIEW
WINGMAN
COMBAT SOUNDS
VARIABLE WEATHER
FIXED TARGETS
SAM SITE AAA
MOVING TARGETS
•TANKS •TRUCKS
WINGMAN
DETAILED TERRAIN
ADV FIGHTER STATION
•SENSOR DISPLAYS•ARMAMENT-GUNS-BOMBS-MISSILES•G-CUES•NAV SYST
• FIRE CONTROLS• MISSILES• GUNS•G-CUES

RECENT ACCOMPLISHMENTS

- TRANSONIC AIRCRAFT TECHNOLOGY
- PROPULSIVE LIFT ENHANCEMENT
- FLOW FIELD COMPUTATION FOR PRACTICAL THREE-DIMENSIONAL COMPONENTS

TECHNICAL GAPS AND FUTURE OBJECTIVES

- FORWARD SWEPT WING AERODYNAMICS
- ANALYSIS OF UNCONVENTIONAL FORCES ON FLIGHT VEHICLES
- EFFECTIVE STOL AERO/PROPULSIVE DESIGNS
- AERO-CONFIGURED MISSILES
- FAVORABLE INTERFERENCE CONFIGURATIONS
- DESIGN CRITERIA FOR LOW RCS DESIGNS
- PROPULSIVE LIFT/INTEGRATED EXHAUST SYSTEMS
- INCREASED SURVIVABILITY TECHNOLOGY

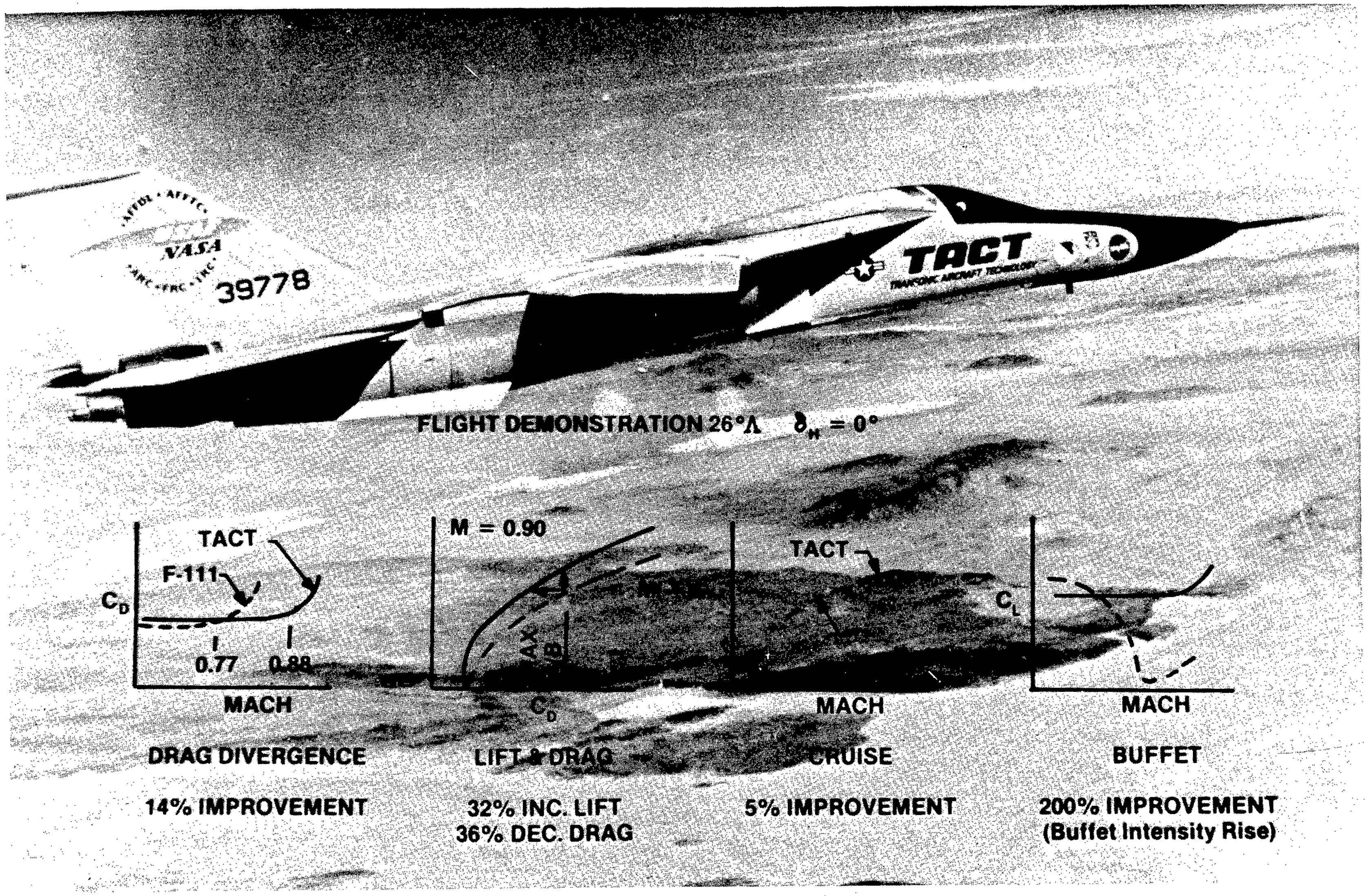


FIGURE 16. TRANSONIC AIRCRAFT TECHNOLOGY

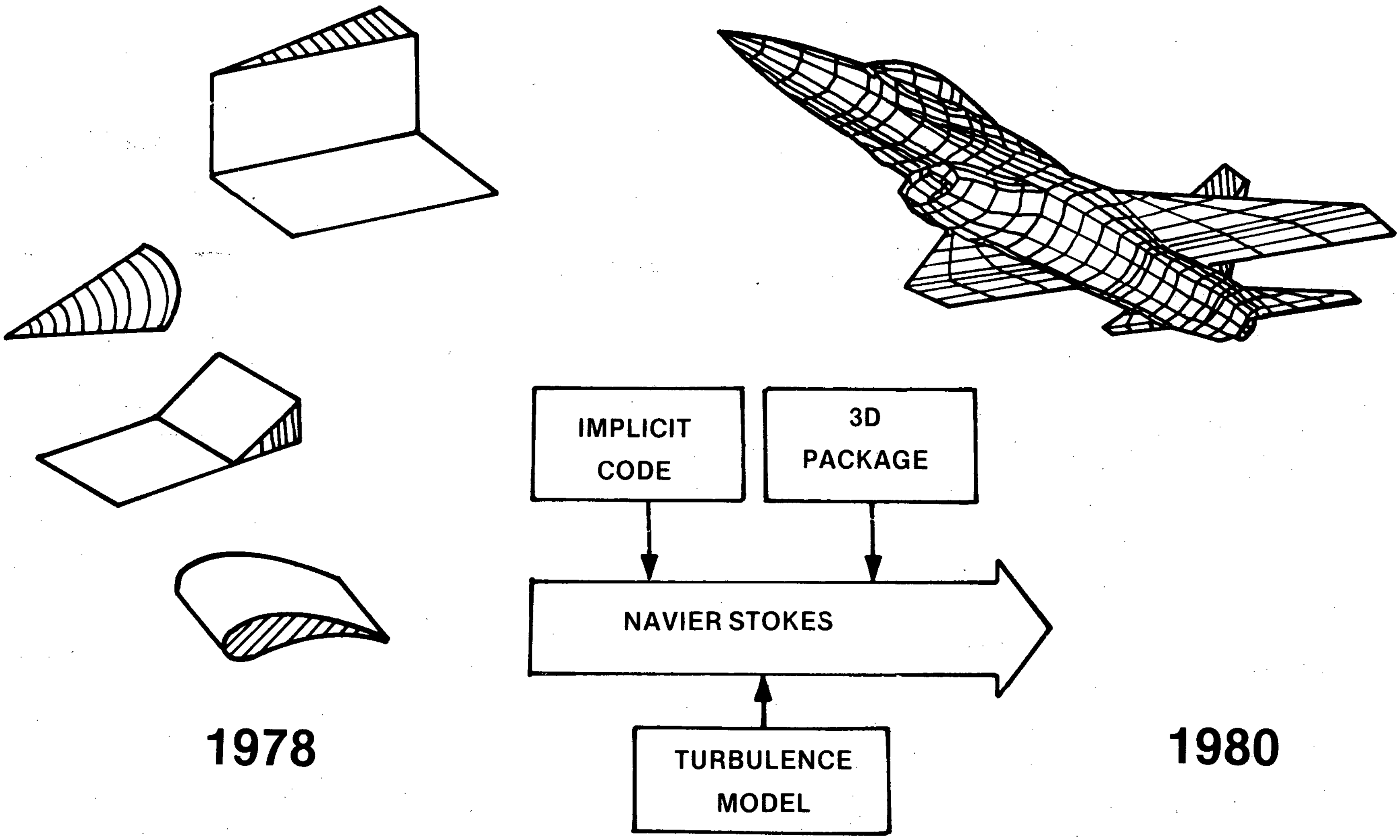
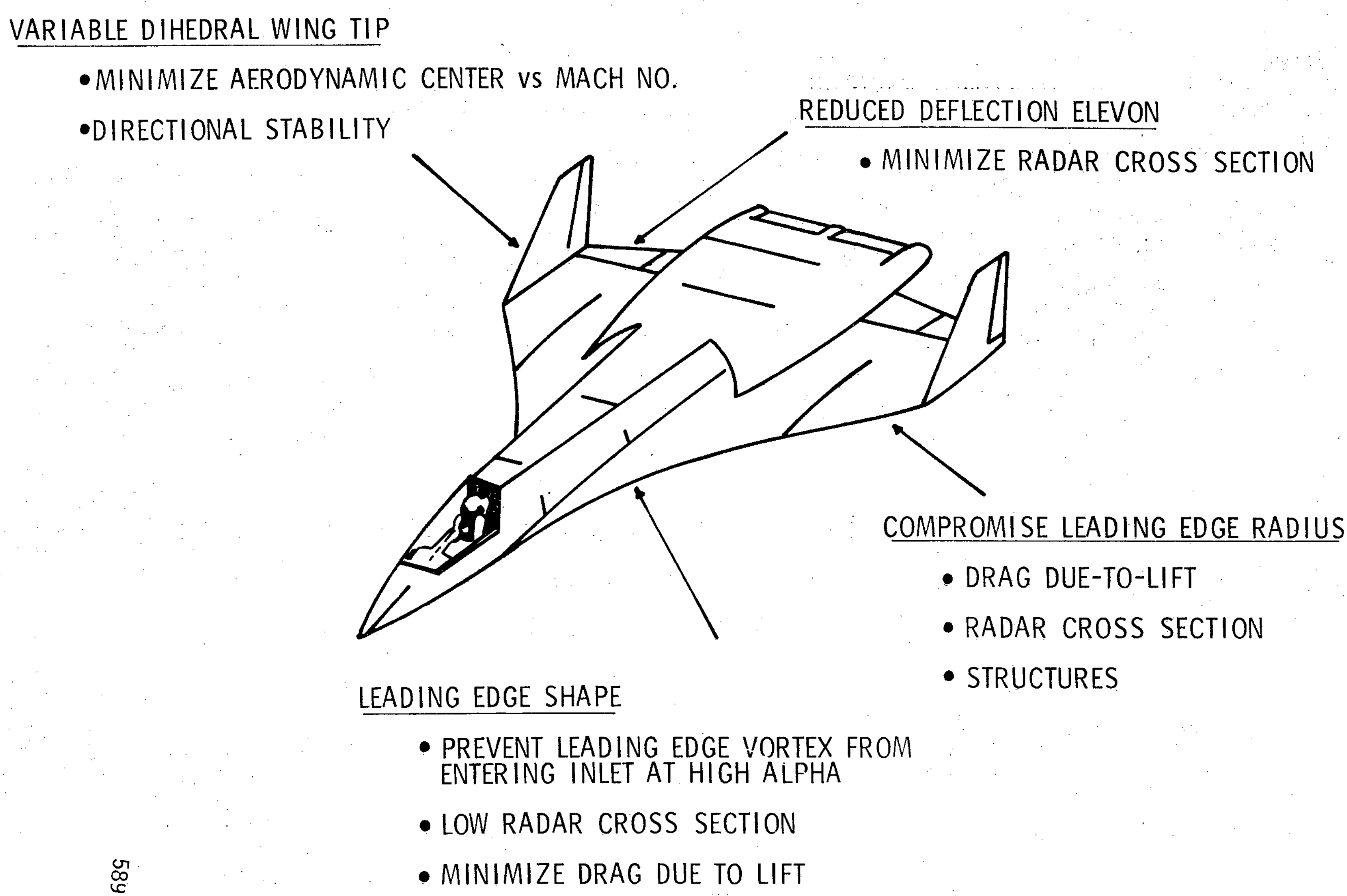
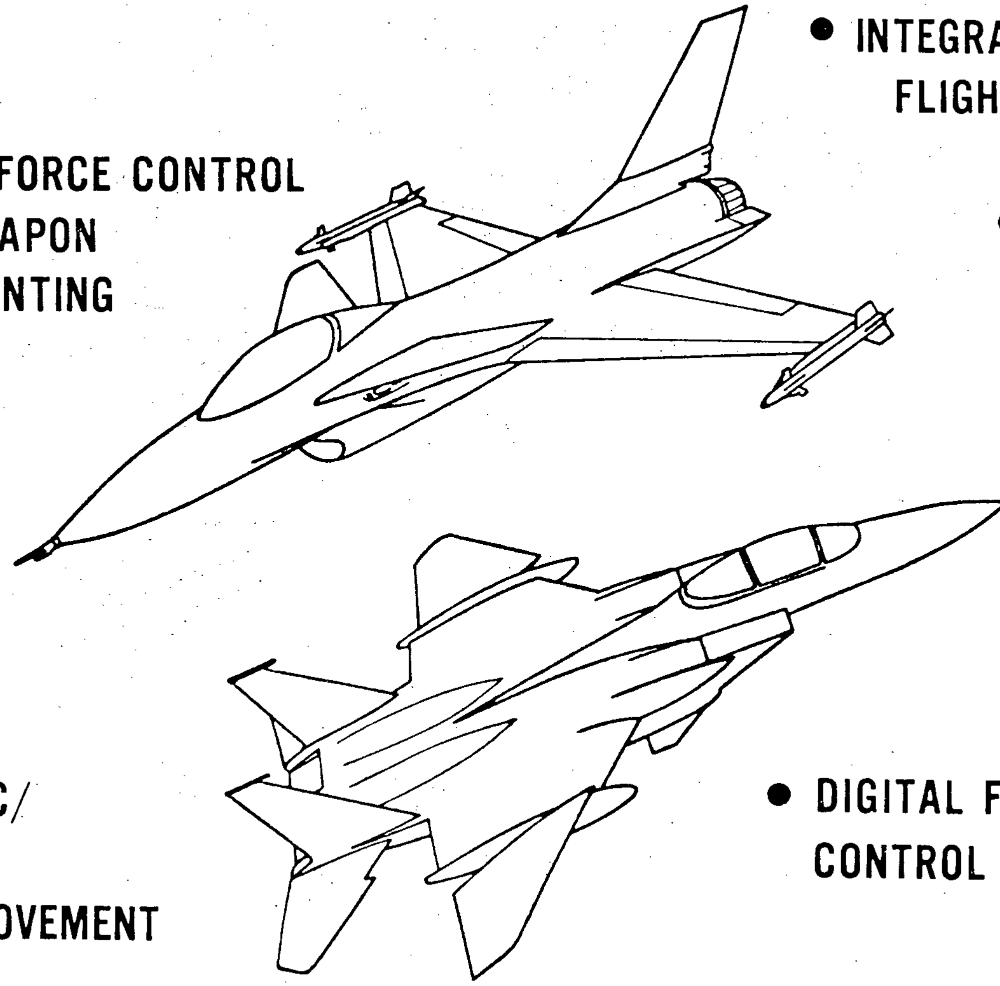


FIGURE 17. FLOW FIELD COMPUTATION FOR PRACTICAL THREE-DIMENSIONAL COMPONENTS



VARIABLE DIHEDRAL WING TIP
• MINIMIZE AERODYNAMIC CENTER vs MACH NO.
• DIRECTIONAL STABILITY
REDUCED DEFLECTION ELEVON
• MINIMIZE RADAR CROSS SECTION
COMPROMISE LEADING EDGE RADIUS
• DRAG DUE-TO-LIFT
• RADAR CROSS SECTION
• STRUCTURES
LEADING EDGE SHAPE
• PREVENT LEADING EDGE VORTEX FROM ENTERING INLET AT HIGH ALPHA
• LOW RADAR CROSS SECTION
• MINIMIZE DRAG DUE TO LIFT

- DIRECT FORCE CONTROL AND WEAPON LINE POINTING
- INTEGRATED FIRE AND FLIGHT CONTROL
- PILOT/VEHICLE INTERFACE ADVANCEMENTS



- AERODYNAMIC/ STRUCTURAL DESIGN IMPROVEMENT
- DIGITAL FLIGHT CONTROL SYSTEM

Biographical Sketch

Colonel George F. Cudahy graduated from the United States Naval Academy and was commissioned in the United States Air Force in June 1957. In June 1959, he completed Fighter Interceptor Advanced Pilot training and was assigned to the 513th Fighter Interceptor Squadron, Phalsbourge, France, where he Flew F-86 aircraft. From December 1959 to July 1963 he flew F-102 aircraft with the 496th FIS at Hahn AFB in Germany.

Colonel Cudahy was then assigned to the Air Force Institute of Technology (AFIT) and received a Master of Science Degree in Astronuatical Engineering in August 1965. After AFIT, he was assigned to Holloman AFB where he flew the F-106, F-100, and T-33 aircraft. Upon leaving Holloman AFB in December 1968, he was upgraded in the F-4D aircraft and flew 186 combat missions with the 555th Tactical Fighter Squadron, Udorn, Thailand. He was awarded the DFC with 1st Oak Leaf Cluster, the Bronze Star Medal and the Air Medal with 1st through 11th Oak Leaf Cluster. From December 1969 to December 1970, he was Chief of Flight Operations in the AFPRO office at Hughes Aircraft Company, Culver City, California. He then went to the AEC Laboratory at Los Alamos, New Mexico, as a Research Associate.

In June 1972, Colonel Cudahy returned to AFIT and received a Ph.D. Degree in Aerospace Engineering in September 1976. In September 1973, he was assigned to the Air Force Flight Dynamics Laboratory (AFFDL) as the Deputy for Laser Application. From September 1974 until August 1975, he was the Chief of the Flight Control Division in the AFFDL. He then served in the Air Force Avionics Laboratory as Chief of the Systems Avionics Division until June 1976.

In May 1977, he completed the Air War College at Maxwell AFB, Alabama, as a distinguished graduate and on 1 July 1977 was assigned to his present position as Commander of the Air Force Flight Dynamics Laboratory. Col Cudahy manages a Laboratory with a staff of 1000 and a multi-million dollar budget for conducting research in the flight dynamics technologies.

A FUNCTIONAL AREA REVIEW (FAR) OF NAVY FLIGHT DYNAMICS

BY

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Introduction

Some of the material appearing in this functional area review has been abstracted from the Navy Technical Strategy for the Air Vehicle Exploratory Development Program.(1) The strategy serves as a rational medium within which a description of the flight dynamics technology program can convey a meaningful relationship to major driving factors. An identification of the Navy's role relative to the DOD, NASA, and industrial technical community will be followed by the scope and objectives of the overall program. In turn, the sub-program structure and major thrusts will be presented for each of the several technical areas which comprise flight dynamics. The review will conclude with a discussion of the long-range schedule of technology milestones relative to the Navy's planned aircraft development programs and a synopsis of major accomplishments achieved.

Scope

The U. S. Navy recognizes the need to devote a limited amount of resources to the near-term solution of pressing fleet problems requiring the application of new technology. Figure 1 depicts the distribution of aero R&D among the various governmental agencies and indicates how the flow of technology base development supports generic classes of aircraft vehicles.

Among the various government agencies, NASA and the Air Force tend to dominate the aeronautical research and technology with 30 percent and 29 percent, respectively. The Army accounts for approximately 26 percent and the Navy about 15 percent.

To a large extent, the U. S. Navy depends heavily upon NASA and the Air Force to develop the technology base for large aircraft, subsonic and STOL, as well as for highly maneuverable, high speed tactical aircraft. Thus, for example, the U. S. Navy has minimal programs in STOL aerodynamics and transonic/supersonic aerodynamics. We maintain a minimal program in those areas in order to maintain technical competence and to explore concepts of special interest to the Navy technical community such as the Circulation Control Wing.

In like fashion, the Navy looks to the Army for much of its technology for advanced helicopters or rotary wing aircraft. We confine our resources to areas of particular interest to the Navy, i.e., high speed helicopters, or to concepts of unique potential developed within the Navy laboratory community such as the Circulation Control Rotor. In other cases, where

(1) Koven, W., De Crescente, C., et al., "FY-79 Air Vehicle Technical Strategy," 19 April 1978.

the Navy has a pressing need not being covered adequately by either the Army or NASA, we will take the initiative. The advanced hybrid composite rotor bleade for a H-3 type helicopter is a good example of this kind of activity.

A need is also recognized to devote substantial longer-term assets to overcome known and projected deficiencies in planned Navy aircraft of the future. This includes updated versions of current aircraft as well as entirely new aircraft systems. Additionally, the program strategy acknowledges the importance of "technology push" and actively seeks to exploit new technology to provide major increases in Navy capability. Finally, the program strategy is paced by a determination to meet the windows of application opportunity. In this respect it must be timely with regard to follow-on Advanced Development activities as well as the planned fleet introduction of new major aircraft systems and must encourage acceptance by the aircraft industry, by Navy material acquisition elements and fleet operating forces. Emphasis is placed upon those developments promising high payoff in the time frame of interest.

The U. S. Navy structures its flight dynamics supportive technology areas mainly through two major exploratory development programs: (1) Air Vehicle and (2) Life Support. In addition, research (6.1) and advanced development (6.3 and 6.3a) programs are explicity planned and integrated to effect productive technology transfer to and from, respectively, the 6.2 exploratory development programs. In order to conveniently align the U. S. Navy's flight dynamics program with that of the U. S. Air Force the following supportive technical disciplines have been selected for this functional area review:

AIRCRAFT STRUCTURES,

AERODYNAMICS/STABILITY AND CONTROL,

ESCAPE SYSTEMS AND CRASH SAFETY,

ADVANCED SYSTEMS/CONCEPTS,

AUTO FLIGHT CONTROLS AND DISPLAYS,

MECHANICAL/HYDRAULIC/ELECTRICAL FLIGHT CONTROL AND POWER SYSTEMS.

Figure 2 describes the technology breakdown for Navy Flight Dynamics and its relationship to the parent technology block programs and to the research and advanced development programs as well.

Overall Objectives

The Flight Dynamics Technical strategy recognizes three major challenges. First, it must provide the technology base as well as various systems alternatives to meet projected future air systems vehicle requirements. These requirements are usually expressed in terms of performance parameters and other parameters such as survivability, reliability, maintainability, affordability, or supportability. Secondly, it must be forward looking and alert to exploring the feasiblity and practicability of new concepts and technologies. It must be prepared to expend high risk money to achieve large improvements in naval air capability and/or to seek new ways of performing naval air missions, and maintenance concepts. Finally, the strategy must be aware of and prepared to accept a wide variety of generic fleet problems whose relatively near-term solutions depend heavily upon a vigorous exploitation of emerging exploratory development concepts.

In the main, advances in technologies related to air vehicles are usually non-vehicle-specific; that is they usually have application to all air vehicles. Thus, improvements in lift and drag, reductions in structural weight, increases in safe egression envelope, etc. are often applicable to VSTOL aircraft as well as to large VP type aircraft.

A review of future naval aviation planning reveals the following:

- There is a reasonable probability that one or more types of V/STOL aircraft will be developed within a 20 to 30 year timeframe.

- In the late 1990's large vehicles with long endurance and long range may be required to fulfill the land based VP mission currently being met by the P-3 aircraft.

- The problem of accomplishing the 1990-2000 year tactical medium attack mission has not yet been resolved. It is not clear that a credible V/STOL Type "B" aircraft can fill this need. There is a probability that an all V/STOL aircraft Navy is impractical and the CTOL, carrier-based aircraft, will be required in the future.

The Navy plans to exploit the possibilities offered by V/STOL by employing them in large numbers and a wide variety of Navy ships. In this manner, the Navy hopes to find new ways to put air power to sea and to provide the ability to destroy or neutralize the enemy at long range. Because of uncertainties regarding the future posture of the

carrier Navy, continuing efforts will be directed to upgrading the capabilities of carrier based CTOL aircraft. Total dependence at this time on V/STOL aircraft is to be avoided.

To provide a survivable first-pass "kill" capability for projection-of-power-ashore forces, the Navy plans to utilize long range standoff weapons to avoid exposure over enemy defenses. In addition, the Navy is counting heavily on technology to provide increased survivability and reduced vulnerability to enemy fire.

To cope with possible loss of land bases for long range ASW operations, a potential solution is to increase significantly the range and endurance capabilities of VP type aircraft or to develop new concepts which reduce our dependence on insecure foreign land bases.

In order to decrease cost of acquisition and support and to improve fleet readiness to an acceptable level, current thinking is that the Navy must: decrease the number of aircraft types through the use of flexible multi-mission-capable aircraft, maximize subsystem commonality through greater standardization and modularity, and simplify system design and logistic support.

These broad objectives, derived from the Navy's near-term and long-range requirements, form the basis for establishing the specific objectives or major thrusts within each of the technology areas comprising Flight Dynamics. The major thrusts can be interpreted as satisfying the following:

- The need to reduce airframe and other aircraft subsystem weight;
- The need to extend the useful service life of Navy/Marine Corps aircraft;
- The requirement to provide aircraft and aircraft subsystems high reliability and easy repairability; and
- The demand to operate safely from austere land bases as well as a wide variety of air capable ships under day/night all weather conditions.

Figure 3 summarizes the flow of considerations which lead to the development of tangible objectives for the Navy's Flight Dynamics program.

Major Thrusts and Program Description

Each technology area is described below in terms of (1) explicit technical disciplines and scope of application, (2) specific major thrust(s) being addressed, and (3) program structure and content.

Aircraft Structures - The aircraft structures technology program encompasses the identification and evaluation of structural design concepts, advanced material applications and life estimation methods to provide options in accordance with system requirements, for structures exhibiting the proper mix of efficiency, low cost and service life durability.

Major Thrusts -

- Reduce A/C Structural Weight 15-20 Percent.

  Lightweight airframe structures through the use of composite materials to reduce airframe weight and to reduce corrosion maintenance by 50 percent.

- Extend A/C Service Life by 50 Percent.

  A structural integrity program utilizing advanced methods of tracking airframe structural life usage to achieve and increase in aircraft service life.

Program Structure and Rationale - Figure 4 provides a graphic illustration of the principal work areas in aircraft structures and how these relate to the major objectives (thrusts) given above. The programs represent a good balance of 6.1, 6.2, and 6.3+ support and concentrate on developing a sound composite technology base along with an equally effective base upon which to develop a credible service life extension methodology and implementation.

Future Navy fighter-type aircraft require airframes that are lighter in weight, lower in cost, longer lasting, and easier to maintain than conventional aircraft construction. Major design constraints on naval aircraft are takeoff and landing gross weights for carrier operations; these often limit useful load and/or operational flight profiles. In addition, the weight-sensitive advanced V/STOL fighter will need 25 to 30 percent reduction in structural weight to approach the payload/range envelope of conventional aircraft. The focus of the structures technology program is matching the major opportunities with the areas of high leverage. Most significant is the opportunity afforded by fiber reinforced composite materials to economically gain near revolutionary improvements in structural efficiency (15 to 30 percent weight savings on a component basis) together with improvements in fatigue and corrosion (for organic matrix materials) resistance.

Beyond material improvements, advancement in the structural life/durability area rests heavily upon an improved capability to predict (for design) and economically measure (for individual aircraft life tracking) aircraft service load histories. Operational structural integrity and structural fatigue life computation procedure in the Navy are currently based upon a safe life philosophy. In service safe-life philosophy is implemented by a fleet management program based upon tail number tracking.

Actual service experience determined by vertical acceleration (N_Z) counts is converted to damage ratios (life expended/life available), using cumulative damage analysis with a scatter factor of two. The conservative assumptions (severe use spectrum and double life requirement) in this approach are necessary to account for analysis uncertainties, fatigue data scatter and the inaccuracies inherent in the procedures by which N_Z data are converted to internal stress. Because damage ratios cannot be related to actual structural degradation, this conservatism could lead to premature repair, replacement and retirement of fleet aircraft with all the associated cost penalties. Current life-prediction methods are inaccurate and may lead to inadequate designs or, in service, may lead to premature airframe retirement. This view is given some credibility by a number of successful service life extension programs in which full scale fatigue tests have dmonstrated the potential of appreciable life extension.

An opportunity exists to bring together recent developments in instrumentation, computer analysis, simulation and data reduction, fatigue/fracture theory and in-service damage assessment methods to provide a vastly upgraded structural life management/integrity assurance process. The results of this process in combination with advanced material, promise inherently more durable/longer life structures and a better capability to monitor actual service life expenditure (estimates of which can weigh heavily on aircraft grounding, retirement, and rework actions although current practice may, in some cases, be conservative by a factor of five or more).

Aerodynamics/Stability and Control - The basic disciplines included in this technology are: aerodynamics (theoretical and empirical) stability/control, flying qualities, performance, and the various interfaces between the airframe and propulsion system as well as the interactions among the man, the displays, the stability augmentation systems and the basic air vehicle.

Major Thrusts -

- Superior Flying Qualities in support of all weather, V/STOL operations.
- Minimize out-of-control aircraft losses and improve combat maneuverability.
- Improve transonic cruise efficiency.
- Maintain low carrier landing speeds.
- Improve accuracy of aerodynamic estimation methods.

Program Structure and Content - Figures 5 and 6 describe the breakdown of principal tasks in this technology area. For convenience the work has been divided into V/STOL and CTOL categories, respectively.

The urgent requirement to provide technology for the development of efficient high performance V/STOL aircraft concepts and the need to provide aircraft flying qualities adequate to accomplish safe, all weather takeoff/landing operations dominate this area. Key problems are: ground effect and adverse down-wash-induced effects; high thrust losses associated with hover/transition control; inadequate hover/transition control power; hover/high speed design compromises; flying qualities design criteria as affected by ship motion, turbulence, displays, and landing aids. In addition, numerous problems exist with respect to wind tunnel and flight test measurement techniques.

The severity of the V/STOL landing problem can be seen from the fact that the AV-8A aircraft are restricted to daytime VFR flight conditions and operations in calm seas only. Those limitations cause a 15 percent reduction in aircraft operational readiness averaged over a worldwide deployment. Our objective is to develop the technology to allow safe operations under conditions of zero ceiling, 700 feet visibility up to and including sea state 5 (15 foot waves) conditions. Achievement of these goals will result in inoperable weather conditions occurring less than two percent of the time.

Through the use of advanced moving base simulators as well as inflight simulators such as the variable stability X-22 V/STOL aircraft, through the use of large scale V/STOL wind tunnels, through the exploitation of advanced control system and display technologies and with the assistance of improved techniques for predicting downwash flow field characteristics, it is expected that objectives can be achieved.

With respect to CTOL aircraft one of the main concerns to the Navy is the development of improved high lift systems to reduce landing speeds. A reassessment of generically different high-lift systems is being performed to judge the relative performance gains and design penalties associated with blown (externally and internally) and passive (mechanical) systems. Upper surface and combined surface externally blown systems appear to be reasonable candidates for more intensive development. Internally blown systems such as circulation control and augmented jet flap are now under active development efforts by the Navy and NASA respectively. The A-6 CC Wing flight demonstrator program is an excellent example of the potential of such advanced high lift systems. A 50 percent improvement in $C_{L_{TRIM}}$ is expected at a modest blowing coefficient of .05 (in comparison with the standard A-6A 30° degree flap configuration). This translates into a possible 15 percent reduction in landing speed and is significant.

The improvement of high subsonic/transonic cruise efficiency is essential for advanced Navy fighter attack, ASW and COD aircraft. A number of programs are directing attention to the development of new airfoil and wing designs which promise higher L/D and extended drag-rise Mach numbers without attendant degraded off-design performance. Programs ranging from developing the variable camber wing to exploring the dromedary airfoil are addressing this problem area.

A problem area shared by the Air Force and Navy is that of a rising incidence of out-of-control losses. Aircraft departure characteristics are rapidly being better understood through intensive efforts by both services. Over the past two years, new design criteria have evolved based on aircraft open and closed-loop considerations. These advanced criteria are just now entering, under a joint Navy/Air Force program, a manned simulation validation phase. With the intensive current re-examination of MIL F-8785B, the outcome of the present work in this area should prove timely.

Escape Systems and Crash Safety - The disciplines applicable to this technology area include anthropometrics (human systems math modeling), seat propulsion systems, physiology and psychology (in excessive environments), design of flight equipment and test systems engineering.

Major Thrusts -

- Vehicle Escape: Develop expeditious escape systems designed for the character and size of the vehicle, the number of its occupants, the range of available escape routes and the time required for emergency transit and the hazards of the departure environments.

- Vehicle Seating and Restraint: Develop seat-installed, adjustable, quick-release, personnel-restraint systems for stabilizing operator body, head and extremities in high performance vehicles, designed into seats incorporating energy-absorbing/force-attenuating means as may be required for moderating vibration, sustained and oscillatory g-forces and oscillatory impacts, as well as the forces transmitted during survivable crashes.

Program Structure and Content - Figure 7 provides a description of the principal tasks addressing the major thrusts of this technology area.

Advanced Systems/Concepts - Advanced systems concepts formulation encompasses studies and analyses of unique and promising air vehicles that offer a potential impact on future USN/USMC systems; and studies of air vehicle/ship basing compatibility and/or interface problems.

Major Thrust -

- Develop Innovative Air Vehicle Concepts.

  Innovative air vehicle systems concepts having potential for major impact on naval operational doctrine, capabilities and force structure as well as on aircraft effectiveness, cost and availability. This includes concepts such as V/STOL "B" and "C", WIG, LTA, LMNA, X-Wing, seacraft, etc.

Program Structure and Content - Figure 8 depicts the principal tasks which are being directed toward the major thrust of this technology area. Several Navy-initiated advanced air vehicle concepts are currently under investigation primarily to develop a data base sufficient to judge potential advantages.

Operational requirements indicated in the STO's (Scientific and Technology Objectives) and Naval Aviation Plan necessitate the development of aircraft technologies beyond the present state-of-the-art to meet critical needs pertaining to performance, cost effectiveness and readiness. This program is responsive to the technology pull exerted by systems effectiveness projections.

Additionally, in order to assure the timely entrance of new aircraft systems into the fleet, advanced or innovative aircraft concepts must be explored 10 to 15 years prior to the need for these concepts in the performance of future Navy missions against the expected threats of the time period. In this way, aircraft concepts with potential application to future Navy missions are identified at an early stage and can then be subjected to a more timely, intensive, and carefully controlled development.

Therefore, system studies conducted under this element establish technology goals for the various technical disciplines, define the relative worth of different approaches and alternatives, and provide the basis for prioritizing competing technical opportunities.

In addition, system studies define the range of alternatives to be investigated in greater depth and establish the spectrum of feasible air vehicle requirements upon which to focus technology. Further, this effort includes design studies and necessary experimental investigations of advanced vehicle concepts.

VTX design studies (both in-house and contracted) are trending toward a highly versatile, fuel-conservative, variable-stability trainer aircraft with derivative forms potentially suitable for satisfying other major Navy mission roles. The trainer, as currently conceived, will replace both the T-2C and TA-4 aircraft presently being used. Studies have naturally impacted the entire form of the future Navy flight training syllabus and will undoubtedly lead to a more optimum blend of ground-based and in-flight training segments. Preliminary conceptual design technology studies and mission definition are nearing completion. With the approval of the impending MENS (Mission Element Need Statement), the VTX program will enter the design competition phase. The aircraft is planned for introduction to the fleet in 1986+.

Flight Control/Displays and Instrumentation - This technical program covers primary flight controls, stabilization systems, automatic pilot functions, cockpit instruments, and displays. This includes the associated sensors, data processing, and actuators for these systems.

Major Thrusts -

Automatic Flight Controls:

- Superior Flight Controls and Controls Integration of all weather, V/STOL operations.

  Digital Fly-by-Wire flight control systems matched to integrated programmable display systems to provide safe all weather V/STOL terminal area landing capability as well as desired up and away flight characteristics. Digital computation, multiple redundance concepts, modularization, integrated display technology and advanced flight simulation techniques are key elements in attainment of objectives. Reliability improvement of 2 to 1 and weight savings of 20 percent are expected.

Displays and Instrumentation:

- Reduction in Pilot Workload
- Reduction in Component Parts
- Multi-Platform Capability
- All-Weather Capability

Program Structure and Content - Figure 9 shows a schematic of the major advanced development work in this area, i.e., the development of V/STOL Digital automatic flight control systems. Shown also are the principal exploratory development supportive programs which, together with the main effort, are directed at achieving the major thrust in this technology area.

Conventional flight control and display/instrumentation systems are limited by: high vulnerability, complex analog-mechanical control linkages with critical rigging and maintenance needs, inflexible control laws, dedicated "steam gage" instruments, numerous raw data non-integrated displays, hundreds of switching functions, and limited built-in-test capability. These problems result in excessive loss rates, low reliability and availability, inefficient information transfer to the crews, overburdened crews with attendant performance degradation. V/STOL vehicles with their inherent instability in the transition and hovering mode and combination aerodynamic/engine thrust and reaction control requirement, will require increased emphasis on control/display integration.

The operational requirements of aircraft such as V/STOL "A" demand maximum performance from the flight control system. In order to further this goal, a DFCS (Digital Flight Control System) has evolved. The DFCS will consist of two major elements, the DFBW (Digital Fly-By-Wire) system and the DFMS (Digital Flight Management System). The DFBW will control the aircraft through redundant electronic paths with no mechanical links. It will have the capability to sustain two similar failures in the electronics with no appreciable degradation of performance. A backup system is accumulated in the operation of the software. The DFMS portions will emphasize the pilot interface and integration of flight controls and cockpit displays. There exists a number of technical deficiencies required to implement a DFCS; in particular, the realization of a DFMS with full capability demands extensive development.

Flight management and crew system integration concepts will provide capabilities never before present in a military aircraft. The implementation of these concepts is expected to reduce pilot workload and require less panel area. Head-up and multi-function, time-shared displays and controllers will provide data keyed to the flight control mode selected and needed by the pilot to accomplish his immediate objective. Hands-on throttle and stick weapons and mode control will be provided for all tasks requiring rapid access. A simple and effective means of communication between the pilot and computers will be provided. The flight management computer, in addition to the pilot, will perform automatic position fixing, steering and flight plan management. These flight management characteristics are expected to provide a significant step toward enabling the pilot to become a mission-oriented manager rather than a subsystem operator.

Holographic techniques promise to provdie wide angle head-up displays system and helmet mounted displays that will allow the pilot to scan the outside world while receiving critical flight information. The capability is particularly important for operation of V/STOL from non-aviation ships under adverse weather conditions. Color display promise to increase the information flow to the pilot without excessive clutter and confusion.

Mechanical/Hydraulic/Electrical Flight Control Systems - The technology program relating to mechanical/hydraulic/electrical systems includes mechanical power transmission components such as bearings and splines; cargo handling and restraint systems; hydraulic pumps, reservoirs, fluid transmission lines, seals, hoses and connectors; electrical power generation, distribution, and control systems.

Major Thrusts -

- Increase A/C hydraulic system power capacity 50 percent.
- Lightweight (30 percent less weight), low volume hydraulic systems utilizing fluids at very high operating pressures, advanced tubing and seal materials and new concepts in connectors and hoses.
- Increase A/C electrical system power capacity by 100 percent.
- Lightweight (45 percent weight reduction), extremely reliable electrical systems exploiting new high voltage DC generator concepts, solid sate switching devices, central control concepts and multiplexing.

Program Structure and Content - Presented in Figures 10, 11, and 12 are the principal task addressing the major thrusts in mechanical/hydraulic/electrical flight control and power systems.

Recent studies have indicated that auxiliary power system requirements in Navy aircraft will increase substantially in the future. Increases in hydraulic power requirements are projected to be approximately 50 percent and electrical power requirements are expected to double. This trend will produce excessively heavy/bulky systems of low reliability and high maintenance requirements unless significant departures from current design practices are followed. In addition, current mechanical components, particularly in rotary wing aircraft transmissions, continue to be a serious maintenance burden. In the sum, auxiliary power system components account for over 50 percent of unscheduled repair times for rotary wing aircraft and 30 percent for large combat fighters. The twin specters of high weight and unreliability while damaging to current Navy aircraft will be intolerable for a V/STOL Navy of the future.

Mechanical/Hydraulic Systems - Auxiliary hydraulic mechanical equipment must take advantage of technological improvements in materials and manufacturing processes to maintain aircrew safety and equipment survivability, reduce aircraft downtime and improve performance capability.

An improved hydraulic system is needed to overcome potential weight and space problems, improve maintenance/replacement access capability and reduce current hydraulic systems susceptability to leakage, contamination and projectile damage. This improved system will be acheived by applying the latest technologies related to hydraulic components, fluids, connectros, seals and filtration systems.

In attempting to prioritize research and development tasks that will benefit the fleet, the projected cost effectiveness of the proposed effort becomes a major consideration. While a proposed system or methodology may be very desirable, what is total impact expense going to be to the fleet? A method is needed to accurately predict the economic benefits of a program entering 6.3 advanced development.

Electrical Systems - Current 400 hertz constant frequency electric power systems rely on hydromechanical constant speed drives to convert the variable engine shaft speed to the constant speed required for constant frequency power generation. These devices have a long history of high cost and poor reliability. Extensive development effort to improve reliability have not produced results commensurate with cost. Recent developments in solid state cycloconverters may improve reliability at the cost of complexity and is considered as one alternative to the development efforts on the High Voltage DC system.

To meet the requirements for future aircraft such as MPA and V/STOL significant advanced in system reliability, weight reduction and system integration must be accomplished. The technology program in electrical systems addresses these requirements through a consideration of both internal and external system impact. Internal considerations cover the power generation and distribution system and component hardware. External considerations cover the system impact on the aircraft engines, accessories, avionics and environmental control system.

The state-of-the-art in solid state technology is frequently the pace-setter for system breakthrough. Solid state devices for power switching are available for many applications and are being developed where gaps exist. Environmental conditions with unconditioned compartments and within machinery must be addressed to either lower temperature to permit use of available technology in micro circuits or to develop devices which can operate at temperature up to 150°C.

The alternative to the High Voltage DC system is to use one of the solid state approaches to a constant frequency, cycloconverter or DC link. Both would improve electrical system reliability, but the High Voltage DC system would be more reliable and more cost effective because it is less complex and requires fewer components.

Also, they do not offer the weight reduction advantage for avionic systems which DC does.

Broad Schedule - Figure 13 portrays, in priority order, the point in time when selected technologies will be available for transition to 6.3 advanced development. This condensed "time-line diagram" relates the occurrence of transition milestones to the known "windows" for introduction of newly developed Navy aircraft to the fleet.

The planning of advanced technology development, which to a large extent is based upon the expectation of new problems offered by future specific air vehicles, cannot always be accurately paced to changing objectives and re-emphases which alter IOC dates. Nevertheless, a good balance of programs which have broader vehicle application along with those which are highly vehicle-specific is reflected in the Navy's Flight Dynamics Program.

Major Accomplishments

Several Accomplishments of significance are listed in Figure 14. These major milestones were realized through a methodical exploratory development process which, for the most part, matured to an advanced

development level. The relevance of these achievements to the solution of current fleet problems and to the satisfaction of projected naval aviation needs has resulted from constant direction of work toward specific major thrusts in each respective technology area. The accomplishments represent a wide range of technology development; from the advanced structures effort which led to the AV-8B composite wing to the innovative application of circulation control in advancing the state-of-the-art in the design of helicopter vehicles and high-lift systems.

Some of the papers scheduled to be presented in the Flight Dynamics Session deal directly with a number of these accomplishment areas. All of the papers, however, represent areas of significant interest to the Navy and expectantly to the entire Research and Development community as well.

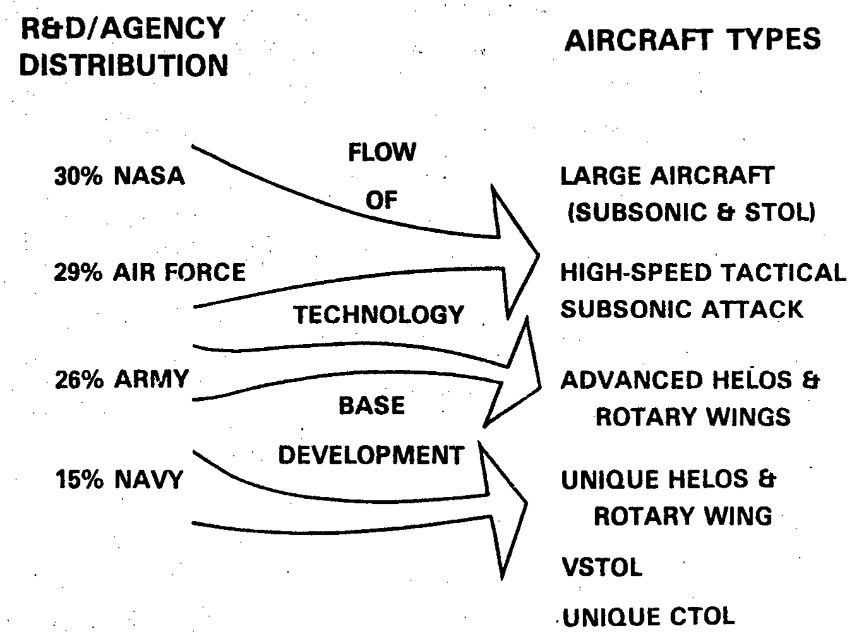

FIGURE 1. U. S. NAVY'S ROLE IN AIRCRAFT FLIGHT DYNAMICS

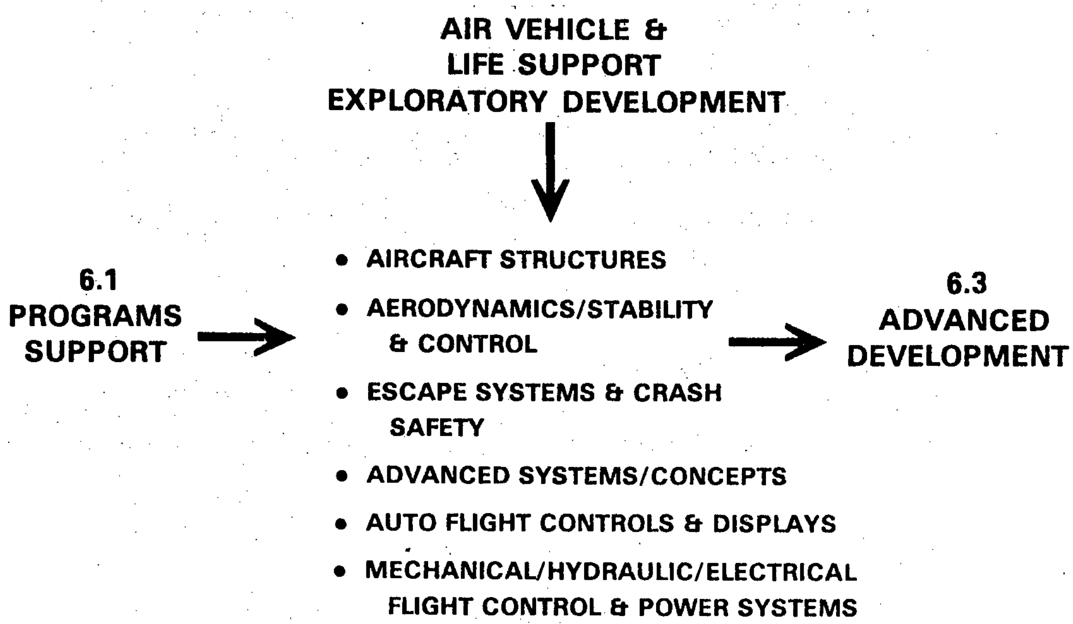

FIGURE 2. NAVY FLIGHT DYNAMICS: TECHNOLOGY BREAKDOWN

GUIDANCE & REQUIREMENTS

- NAVAL AVIATION PLAN
- SCIENTIFIC & TECHNICAL OBJECTIVES

MAJOR CHALLENGES

- DEVELOP A SOUND TECHNOLOGY BASE & SYSTEMS ALTERNATIVES
- EXPLORE NEW CONCEPTS & NEW TECHNOLOGIES
- SOLVE A VARIETY OF FLEET PROBLEMS

TANGIBLE OBJECTIVES

- REDUCE AIRCRAFT WEIGHT
- IMPROVE SAFETY & EFFECTIVENESS IN SEVERE ENVIRONMENTS
- EXTEND USEFUL SERVICE LIFE
- INCREASE RELIABILITY & REDUCE MAINTAINABILITY

FIGURE 3. DEVELOPING OBJECTIVES

PRINCIPAL TASKS

- AIRFRAME STRUCTURAL MECHANICS
- ADVANCED COMPOSITE STRUCTURES → REDUCE VEHICLE WEIGHT, SIZE AND COST
- COMPOSITE FUSELAGE CRITERIA
- COMPOSITE REPAIR TECHNIQUES
- STRUCTURAL FRACTURE CONTROL → EXTEND USEFUL AIRCRAFT SERVICE LIFE
- STRUCTURAL INTEGRITY METHODOLOGY
- AIRBORNE STRUCTURAL INSTRUMENTATION
- SUPPORT OF CILOP PROGRAMS
- SAFE PROGRAM
- FLIGHT & LANDING LOADS INVESTIGATIONS
- FATIGUE LIFE TRACKING METHODOLOGY
- SERVICE LIFE EXTENSION PROGRAMS

MAJOR THRUSTS

- REDUCE VEHICLE WEIGHT, SIZE AND COST
- EXTEND USEFUL AIRCRAFT SERVICE LIFE

FIGURE 4. PROGRAM STRUCTURE: AIRCRAFT STRUCTURES TECHNOLOGY

PRINCIPAL TASKS

VSTOL AERODYNAMICS

- FINITE ELEMENT ANAL. THRUST AUGMENTER WING
- PROPULSION INDUCED AERODYNAMICS
- STABILITY & CONTROL MANUAL
- HI-SPEED FIGHTER/ATTACK AERO.
- TRANSIENT AERO. OVER MOVING DECK
- RECTANGULAR JET CHARACTERISTICS

VSTOL PROPULSIVE LIFE/CONTROL SYS.

- AUGMENTER PERFORMANCE ESTIMATION
- JET-DIFFUSER AUGMENTER DEVEL.
- AUGMENTER TESTING SCALE EFFECTS
- 3-D FLOW CHARS. TUNNEL TEST
- REACTION CONTROL SYSTEM AUGMENTERS
- VSTOL JET INDUCED WATER SPRAY

VSTOL FLYING QUALITIES

- CRITERIA DEVELOPMENT
- X-22A TERMINAL AREA FLIGHT SIMULATION
- LAUNCH & RECOVERY DYNAMICS
- TRANSITION CORRIDOR ANALYSIS
- HEIGHT CONTROL ANALYSIS
- SHIP-INDUCED TESTING

MAJOR THRUSTS

"SUPERIOR VSTOL FLYING QUALITIES IN SUPPORT OF ALL WEATHER OPERATIONS"

"REDUCE ADVERSE PROPULSIVE-INDUCED AERO. FORCES & MOMENTS"

FIGURE 5. PROGRAM STRUCTURE: AERODYNAMICS/STABILITY & CONTROL TECHNOLOGY – VSTOL

PRINCIPAL TASKS

TRANSONICS

- DROMEDARY FOIL
- PASSIVE LAMINAR AIR FOIL
- VOUGHT VARIABLE CAMBER WING
- ATC TRANSONIC AIR FOIL

FLYING QUALITIES

- HI-α PARAMETER IDENTIFICATION
- DEPARTURE CRITERIA DEVELOP
- NAVY/A.F. MANNED SIMULATION PROGRAM
- ADVANCED CRITERIA DEVELOP
- VECTORED THRUST MANEUVERING

HIGH LIFT AERODYNAMICS

- A-6 CC WING FLT. DEMONSTRATOR
- ADV. HIGH LIFT SYSTEMS - BLOWN & PASSIVE

AERO. PREDICTION METHODS

- SUPERSONIC WAVE DRAG PREDICTION
- 3-D DRAG - WING/BODY/NACELLES
- 3-D POTENTIAL FLOW-CURVILINEAR PANELING

MAJOR THRUSTS

"IMPROVE CRUISE EFFICIENCY"

"MINIMIZE OUT OF CONTROL LOSSES & IMPROVE MANEUVERABILITY"

"MAINTAIN LOW CARRIER LANDING SPEEDS"

"IMPROVE ACCURACY OF AERO. ESTIMATION"

FIGURE 6. PROGRAM STRUCTURE: AERODYNAMICS/STABILITY & CONTROL TECHNOLOGY – CTOL

PRINCIPAL TASKS		MAJOR THRUSTS
• MAXIMUM PERFORMANCE ESCAPE SYSTEM • HELICOPTER ESCAPE & SURVIVAL SYSTEM • ESCAPE PROPULSION SYSTEMS • FIRE-RESISTANT PARACHUTE	→	VEHICLE ESCAPE – INVERTED FROM 50 FT AGL – ZERO/ZERO, 600 KEAS – 5-7 YEAR SERVICE LIFE
• CRASHWORTHY SEATS • PERSONNEL RESTRAINT SYSTEMS	→	VEHICLE SEATING/RESTRAINT – 50% WT. REDUCTION – 50% COST SAVINGS – 48% CRASH ATTENUATION – AUTOMATIC, PASSIVE & FAIL-SAFE RESTRAINT

FIGURE 7. PROGRAM STRUCTURE: ESCAPE SYSTEMS & CRASH SAFETY TECHNOLOGY

PRINCIPAL TASKS		MAJOR THRUST
• VTX CONCEPTUAL DESIGN STUDY • VSTOL FIGHTER/ATTACK CONCEPTS EXPLORATORY STUDY • AERO, CONFIGURATION OPTIMIZATION PGM. • X-WING TECHNOLOGY & CONCEPT DEVELOPMENT • WIG TECHNOLOGY & CONCEPT DEVELOPMENT • CCR TECHNOLOGY & DEMONSTRATOR VEHICLE DEVELOPMENT	→	• DEVELOP INNOVATIVE, ADVANCED AIR VEHICLE CONCEPTS – ENERGY CONSERVATIVE – MULTI–MISSION CAPABILITY – UNIQUE NEW MISSION REQMTS. – NEXT GENERATION REPLACEMENTS

FIGURE 8. PROGRAM STRUCTURE: ADVANCED SYSTEMS/CONCEPTS TECHNOLOGY

PRINCIPAL TASKS

6.2

- ADVANCED ACTUATION SYSTEMS
- NAVTOLAND CONTROL/DISPLAY INTEGRATION
- CRITERIA FOR MIL-C-18244A & MIL-F-18372
- FLT. CONTROL/FLYING QUALITIES VSTOL SIMULATION
- INTEGRATED SENSORY SUBSYSTEM TECHNOLOGY

↓

6.3

A.F./NAVY DIGITAL FLIGHT CONTROL/DISPLAY PROGRAM ("AFTI" FLIGHT DEMONSTRATOR)

→

MAJOR THRUST

"SUPERIOR FLIGHT CONTROLS/ DISPLAYS IN SUPPORT OF ALL-WEATHER VSTOL OPERATIONS"

FIGURE 9. PROGRAM STRUCTURE: AUTO FLIGHT CONTROLS TECHNOLOGY

PRINCIPAL TASKS

6.2

- ADVANCED DISPLAY TECHNOLOGY
- ADVANCED DISPLAY INTEGRATION
- HUMAN FACTORS
- ADVANCED INSTRUMENTATION & SENSORS

↓

6.3

ADVANCED INTEGRATED DISPLAY SYSTEM (AIDS)

→

MAJOR THRUSTS

- REDUCTION PILOT WORKLOAD
- REDUCTION COMPONENT PARTS
- MULTI-PLATFORM CAPABILITY
- ALL-WEATHER CAPABILITY

FIGURE 10. PROGRAM STRUCTURE: DISPLAYS TECHNOLOGY

PRINCIPAL TASKS		MAJOR THRUSTS
• LIGHTWEIGHT HYDRAULIC SYSTEM • HYDRAULIC STORED ENERGY CONCEPTS/DEVICES	→	• INCREASE HYDRAULIC POWER CAPACITY BY 100% • REDUCE HYDRAULIC SYSTEM WEIGHT BY 30% & REDUCE VOLUME
• ADVANCED CARGO/RESTRAINT SYSTEMS • FIRE RESISTANT HYDRAULIC FLUID/SYSTEM • AIRCRAFT FLOTATION SYSTEM • ANTI-SKID BRAKE SYSTEM	→	• INCREASE SAFETY & SURVIVABILITY
• HYDRAULIC DIAGNOSTIC & MONITORING SYSTEM • BELLOWS RESEVOIR DESIGN • HELICOPTER BEARING DEVELOPMENT • ADV. SEALS, CONNECTORS/TUBING & FILTRATION • ADV. PANEL FASTENING	→	• IMPROVE RELIABILITY & MAINTAINABILITY

FIGURE 11. PROGRAM STRUCTURE: MECHANICAL/HYDRAULIC POWER SYSTEMS TECHNOLOGY

PRINCIPAL TASKS	MAJOR THRUSTS			
	REDUCE WEIGHT	IMPROVE R&M	INCREASE POWER CAPACITY	
• 270 VDC SYSTEM — GENERATOR — POWER CONDITIONER — BUS CONTROLLER — EMERGENCY POWER — BRUSHLESS MOTOR	*	*	*	→ ADVANCED AIRCRAFT ELECTRICAL SYSTEM
• HIGH SPEED TURBINE ENGINE GENERATOR	*	*	*	
• OPTICAL CIRCUIT WIRING — SOLID STATE ELECTRICAL DIGITAL DATA BUS	*	*		
• SOLID STATE ELECTRIC COMPONENTS — POWER CONTROLLER — 200 AMP TRANSISTOR — BUS FAULT SENSOR — FLAT BUS	*	*		

FIGURE 12. PROGRAM STRUCTURE: ELECTRICAL POWER SYSTEMS TECHNOLOGY

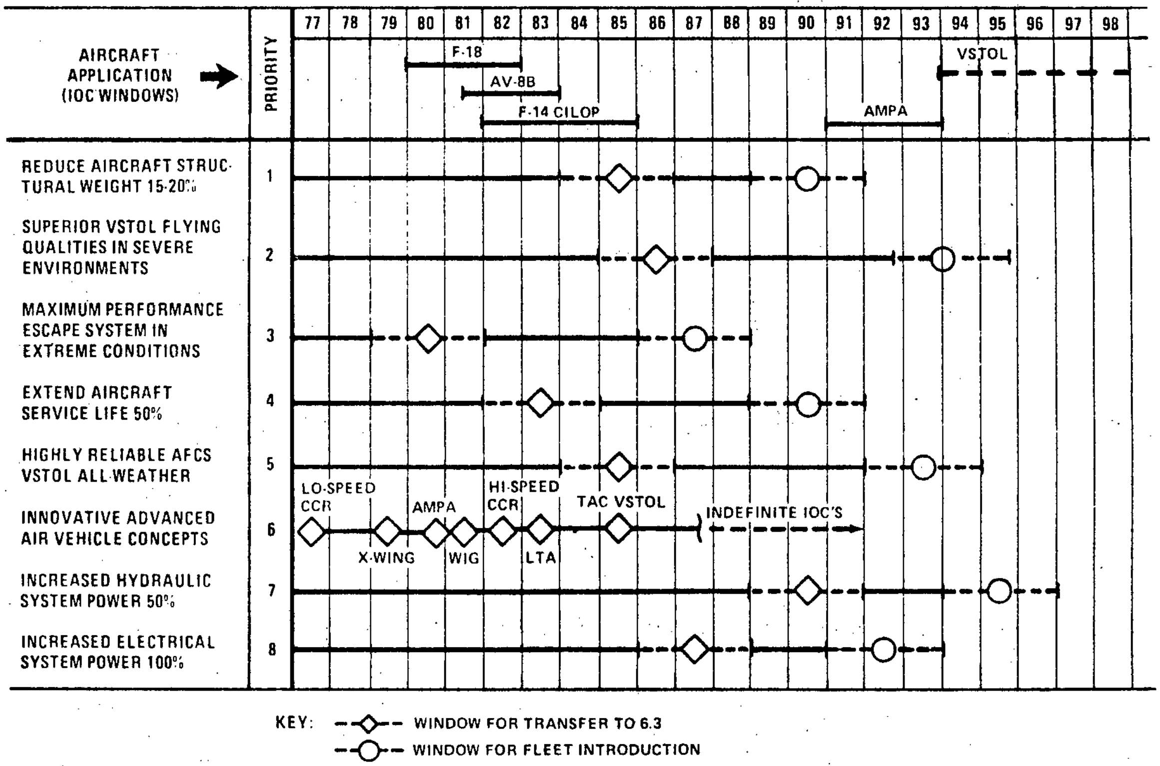

FIGURE 13. SELECTED PRIORITY TECHNICAL OBJECTIVES

- AV-8B COMPOSITE WING 6.2→ 6.3→ FLIGHT TEST
- ADVANCED ELECTRICAL SYSTEM IN 6.3→ LABORATORY DEMONSTRATION
- INTEGRATED SENSOR SYSTEM→ FLIGHT DEMONSTRATION
- HI-PRESSURE/LIGHTWEIGHT HYDRAULICS→ T-2C FLIGHT TEST
- AIDS TECHNOLOGY→ F-18 COCKPIT DESIGN
- ESCAPE SYSTEMS TECHNOLOGY→ 6.3 MPES PROGRAM
- CIRCULATION CONTROL TECHNOLOGY→ 6.3 ↗ HELO FLIGHT DEMO ↘ HI-LIFT FLIGHT DEMO
- X-22A→ SIMULATED AV-8B→ OPTIMUM CONTROLS/DISPLAY
- JET-DIFFUSER AUGMENTER→ AUGMENTATION RATIO=2.1+

FIGURE 14. MAJOR ACCOMPLISHMENTS

AIRCRAFT AFT FUSELAGE SONIC DAMPING

BY

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E. R. Roeser

M. J. Devine

Special Projects Office Code 609

Naval Air Development Center
Warminster, Pennsylvania 18974

Aircraft Aft Fuselage Sonic Damping

Abstract

Acoustical fatigue of structural elements represents an extensive repair/replacement requirement for A-6 aircraft (AFT fuselage). External coatings consisting of an elastomeric urethane and three polysulfides were investigated to determine sonic damping characteristics. The coatings were applied to A-6 aircraft at NAVAIREWORKFAC NORVA. The results of initial vibration measurements showed that significant sonic level reduction was achieved with the coating materials. Operational tests for a period of 20 months also showed significant reduction in crack formation and crack growth. Touch-up formulations for on-site use and a method of inspection without damage to the coatings were demonstrated during the study.

Introduction

Acoustical fatigue of structural elements represents an extensive repair replacement requirement for military aircraft. Several Naval Air Rework Facilities reported a chronic problem of skin cracking resulting from sonic fatigue on the A-6 aircraft. Techniques to resolve such fatigue problems are complicated by (1) the complex relationships between fatigue behavior of built-up structures; this includes design, skin thickness, skin materials, stiffner configuration and the damping of the structure. (2) Then aluminum, the material often specified for aircraft due to weight considerations possesses very low damping characteristics.

The Naval Air Systems Command Atlantic Representative asked the Naval Air Development Center to initiate a program to determine the feasibility of reducing crack formation and crack growth by the application of damping coatings.

Three coatings, an elastomeric polyurethane, a filled polysulfide and nylon-11 applied by plasma spray technique, were selected. The application and processing of the coatings were done in cooperation with the Naval Air Rework Facility at Norfolk.

Discussion

The materials selected were applied to the top, sides and tail hook area of the aft fuselage of four A-6 aircraft. Figure 1 shows the area covered by the coatings. After the aircraft had been treated, the usual exterior finish polyurethane paint was applied. The fuselage skin areas to which the vibration damping material was applied was identified by paint stenciled 3/4 inch long black arrows spaced at 36-inch intervals around the outer perimeter.

In order to determine the effectiveness of the four different coatings, the aircraft were instrumented with the following:

Endevco Accelorometer	P/N 2215D
Sound and Vibration Analyzer	P/N 1564
Preamplifier	P/N 1560-P40

The following A-6 aircraft were used in the study:

BuNo 152915 used as the control, no damping coatings used.

BuNo 151582 coated with "Thiokol" primer and Polysulfide (lead-type, grey) 12-16 mils thick.

BuNo 157027 coated with "Thiokol" primer and Polysulfide (zinc-type, white) 12-16 mils thick.

BuNo 155622 coated with MIL-P 23377B epoxy primer plus an elastomeric polyurethane (Vought Aircraft Corp.), 6-8 mil.

BuNo 155708 first an attempt was made to plasma spray the aircraft with nylon-11. However, because of limited power available compounded by the low temperature in the area made it impossible to apply the nylon-11 properly. The aircraft was then stripped, cleaned, sealed and coated with Mil-P-23377B, 12 mils thick.

Each aircraft was located on a runway and the accelerometers attached at the locations shown in figs. 2 and 3. Vibration data was collected with the two jet engines trimmed and at maximum power. A comparison of the accelerometer data is presented in figures 3 and 4. It can be observed the over the full range of engine noise frequencies that the elastomeric polyurethane and zinc modified polysulfide show significant damping characteristics.

Flight tests were initiated and the results are given.

BuNo 151582 Non-destructive (radiographic) inspection after 146 flight hours showed a single one-half inch crack. Panel replacement was not required. At this time twelve additional A-6 aircraft without damping coatings were examined for purposes of comparison. Each of these aircraft had several large cracks requiring structural repair. After 379.5 hours of operation BuNo 151582 was again inspected along with BuNo 152915 the control aircraft. With these results:

BuNo.	Hours since PAR	Catapult Launches	Skin Cracks	Total Flight Hrs. Since Acceptance
152915 (Control)	248.0	22	27	1480.3
151582 (Coated)	379.5	148	5	2092.7

BuNo 157027 after 150 flight hours showed one small crack. Panel replacement was not required. When inspected after 272 flight hours the one crack had extended to three inches. After ten months no additional cracks were reported.

BuNo 155708 Inspection after 192 flight hours showed no cracks. No reports of cracks after ten months.

BuNo 155622 No cracks reported in eight months flying time. Then was lost in battle.

Since surface coatings on small areas of the fuselage are often damaged by fuel leaks, abrasion, etc., these areas will be more easily corroded thereby accelerating structural damage. These areas may be repaired by cleaning and refinishing with available materials.

The complete procedure is available in Report No. NADC-73069-30 of 24 Oct 1973.

In addition to the in-flight aircraft tests, a test specimen was studied. The test specimen consisted of a section from the aft fuselage of an A-6 aircraft. The specimen was exposed to a high intensity noise environment at the Sonic Fatigue Test Facility at the Wright-Patterson Air Force Base, Dayton, Ohio. The test consisted of four phases. Phase 1 established suitable locations for the installation of strain gauges. Phase 2 produced strain data from the bare uncoated fuselage and Phase 3 established strain values for the coated fuselage. Phase 4 determined the integrity of the coatings during a 50-hour high-intensity sonic fatigue test. Four coatings were selected for the test. Two were polysulfides, one using lead, the other using a zinc catalyst; the third was a polyurethane and the fourth a nylon-11 plasma spray coating. The results of the test show that (1) the external coatings did not reduce the strains in the panels, (2) coating integrity was maintained during the 50-hour high intensity sonic fatigue test (3) and areas having no damping coating showed evidence of crack formation and crack growth. It was observed that crack formation in the aircraft skin did not occur with the nylon-11 coating

Conclusion

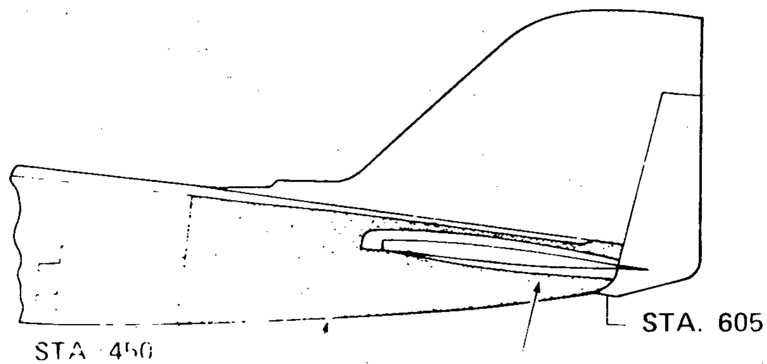
Based on these findings, it is concluded that damping coatings applied to A-6 aircraft fuselage areas can reduce crack formation and crack growth.

ARP/SLP

SONIC FATIGUE

- AIRCRAFT: A-6 (AFT FUSELAGE)
- TEST SITE: NAS, OCEANA
- CFA: NARF, NORFOLK
- RPT: NO. NADC-73069-30
- RESULT: (a) DEMONSTRATED (2 YR.-SERVICE TEST) EFFECTIVENESS OF NEW COATING TECHNIQUE FOR SONIC DAMPING TO DRASTICALLY REDUCE FUSELAGE CRACKING (27 CRACKS, CONTROL VS LESS THAN 5 CRACKS, COATED)

  (b) TECHNIQUE INCLUDED COATING PROCESS, COATING TOUCH-UP, AND COATING INSPECTION TO ELIMINATE NEED FOR COATING REMOVAL FOR CRACK EXAMINATION



SONIC DAMPING TECHNIQUE REDUCES CRACK/GROWTH PROPAGATION

Figure 1. Skin Panel Damping

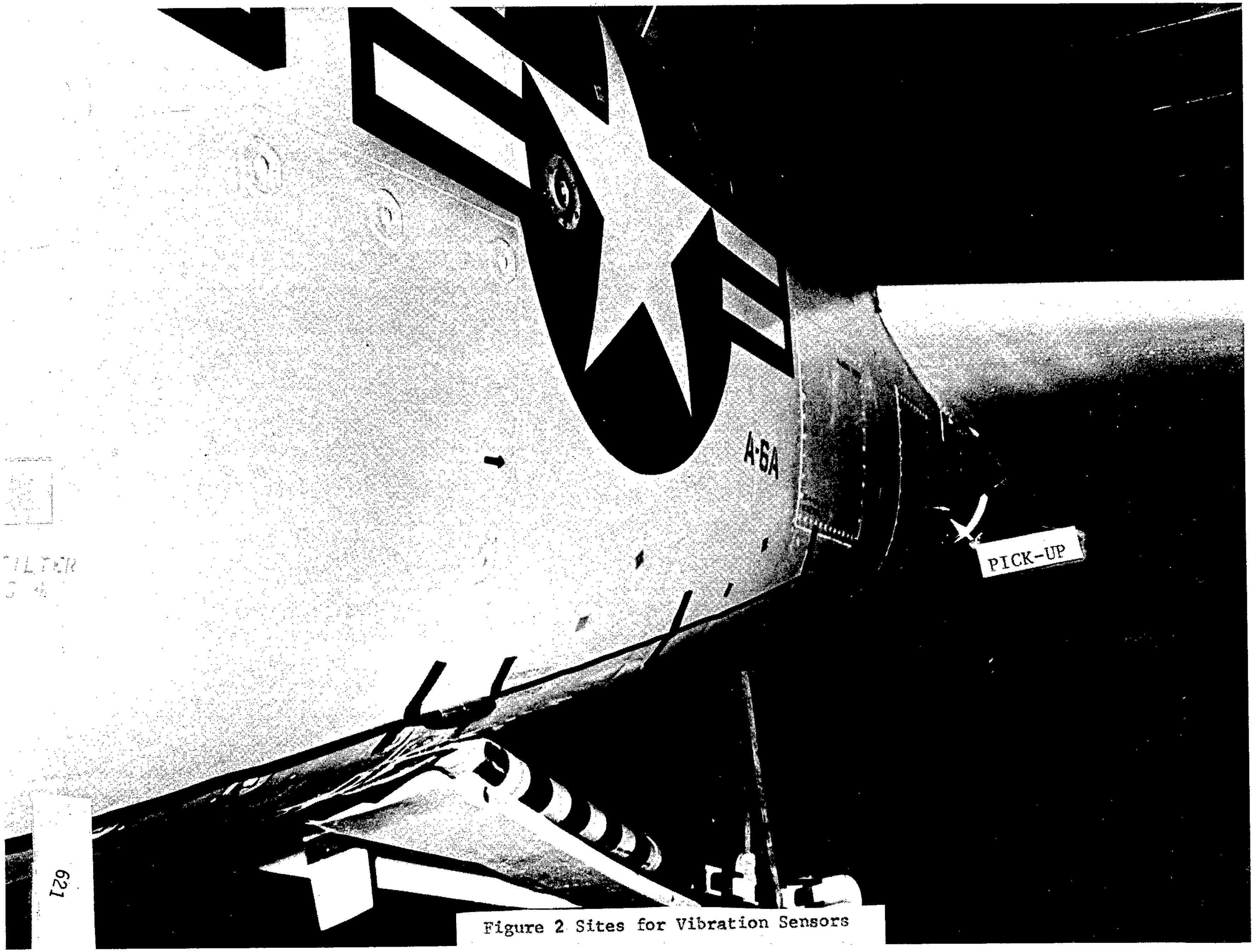


Figure 2 Sites for Vibration Sensors

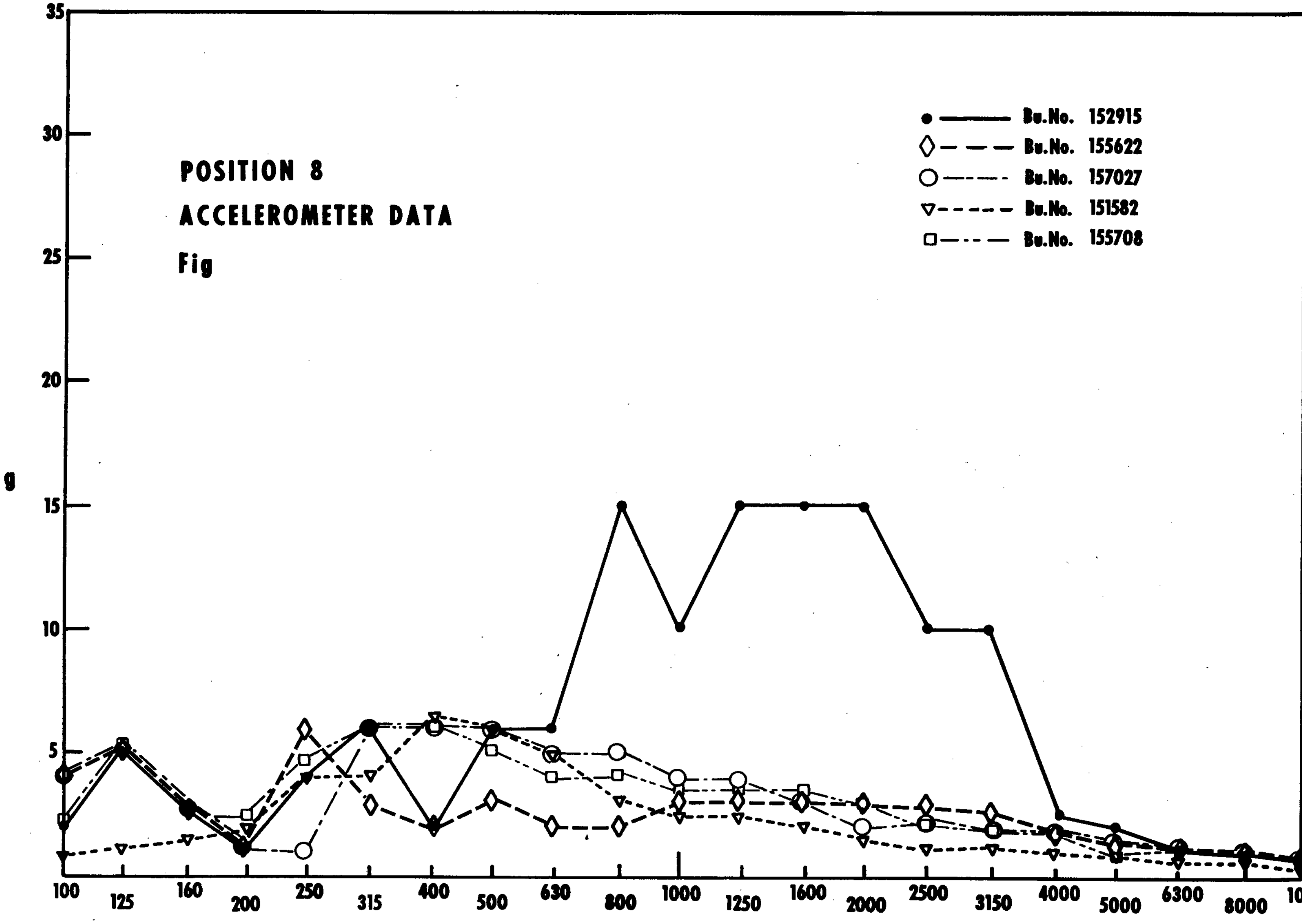


Figure 3. Accelorometer Data

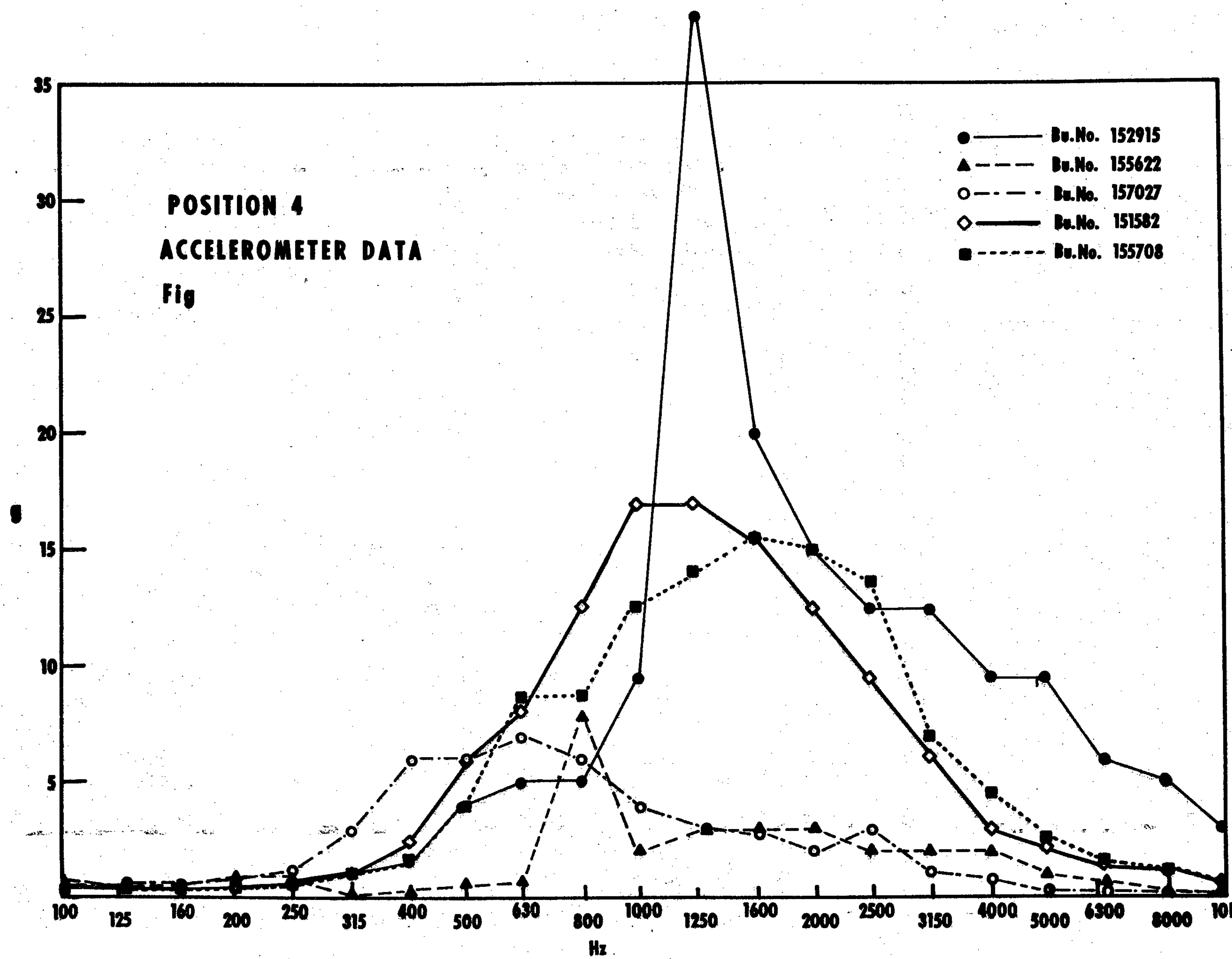


Figure 4. Accelorometer Data

Biographical Sketch

Gladys Hargreaves was born in Tifton, Georgia. She attended Tift College and the University of Georgia where she received a BS degree in 1935 and a MS degree in 1941. She taught science in high school in Ashburn and Georgetown, Georgia and was associated with the U. S. Army Signal Corps, Eastman Corporation, Tufts University Dental School where she taught or was engaged in research. Her career with the U. S. Navy started as a research associate with the Naval Air Engineering Center, Philadelphia, Pennsylvania in 1956, and has worked in many diverse areas such as sonic welding, radome repairs, plasma coatings, fatigue cracking, and composite repairs. Her participation on the panel for the development of a handbook entitled "Adhesive Bonded Aerospace Structure Standardization Repair Handbook" was greatly appreciated both by the U. S. Navy and the USAF. She has published numerous papers and has been a member and held office in many professional societies including the AAAS, AIC, ACS, Graduate Women in Science, APS, and the Microscopy Society. Miss Hargreaves is presently retired but remains active in research continuing her association with ARP/SLP.

Mr. Erwin Roeser is the Deputy Director of the ARP/SLP Project Office, Air Vehicle Technology Department, Naval Air Development Center. The Office is concerned with the technological interface between the Center's Research and Development Activities and the Navy's Aircraft Rework Facilities. The principle areas of concern are materials engineering, structural fatigue, fluid systems, corrosion prevention, and non-destructive inspection. After graduating from Virginia Polytechnic Institute (BSME)in 1950, he worked at the Piasecki Helicopter Corporation, Morton, PA in the Structures Department. In 1958 he joined the Aero Structures Department of the Naval Engineering Center. He was the Branch Head, Flight Loads Environment Branch, conducting surveys in Navy fleet aircraft. Later he became the Division Head, Weapon Systems Engineering Division assigned responsibility for monitoring Navy contractor stress analysis and loads reports for airframe structural design and testing. He is a member of the American Helicopter Society and the American Institute of Astronautics and Aeronautics.

Biographical Sketch

Martin J. Devine received his Bachelor's Degree (1950) in Chemistry from LaSalle College and his Master's Degree (1967) in Chemical Engineering from Villanova University. He attended the Naval War College in Newport, Rhode Island for one year. Mr. Devine spent one year in the Office of Technology Assessment in Washington, D.C. There he conducted a workshop on wear control to achieve product durability. Up to the present Mr. Devine has been the Associate Director of Special Projects in the Aircraft and Crew Systems Directorate at the Naval Air Development Center.

Mr. Devine is co-author of a number of technical papers covering lubricant materials, bearing design and polymer technology and co-inventor of sixteen patents on lubricants and bearings. He is a member of the American Society of Lubrication Engineers, the Coordinating Research Council, and the American Society of Mechanical Engineers, and Research Committee on Lubrication. He is also Technical Advisor to NATO and ASCC technical committees.

ACTIVE CONTROL APPLICATIONS TO WING/STORE FLUTTER SUPPRESSION

BY

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Dale E. Cooley

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Active Control Applications for Wing/Store Flutter Suppression

ABSTRACT

Fighter aircraft are required to carry many combinations of external stores for an efficient combat role. Some combinations can be critical from the standpoint of preventing dangerous aeroelastic instabilities, such as flutter, from occurring within the operational flight envelope. These occurrences can result in extensive and costly store clearance programs and flight envelope degradation. Speed placards are currently the only practical means of preventing many of the store flutter problems. During recent studies, active flutter suppression has been shown to be a promising technique for preventing wing/store flutter. This paper presents the results of an application of an in-house method used to design and analyze aeroservoelastic systems. The analytical procedure involves the use of the FASTOP computer program to obtain the unaugmented flutter results of the aircraft configuration, and the unsteady aerodynamics which cover the frequency and velocity ranges of interest. To analyze the aircraft with the system operating, calculations are conducted in the frequency domain using classical Nyquist stability criterion. The application presented in this paper involves the analyses of a wing/store active flutter suppression wind tunnel model. Flutter suppression tests at transonic speeds were recently completed on a model of a lightweight fighter wing. The leading edge and trailing edge surface control laws that were tested in the tunnel were analyzed, and the resulting stability evaluation are compared with the test data for several different store configurations.

INTRODUCTION

Wing/Store Flutter Problem

The prevention of dangerous aeroelastic instabilities such as flutter is a major consideration in the design of high performance fighter aircraft. Traditionally, safe flight speeds are determined with a 15 percent margin of safety. The prediction and prevention of flutter becomes very difficult and time consuming when the aircraft is required to carry external stores for multirole mission capability. During the aircraft design phases, flutter can generally be prevented through conventional techniques for the initial stores selected for carriage. These techniques include increasing the stiffness, mass ballasting, managing fuel distribution, or moving the store to a different wing location, either spanwise or chordwise. For those configurations defined analytically to be critical in satisfying flutter margins of safety, the effects of these changes are evaluated through detailed wind tunnel tests. In some cases flight flutter tests are required to verify flight safety for a particular store configuration.

External store flutter prevention becomes increasingly more difficult once the aircraft becomes operational and new stores are required to be carried. As different store configurations are carried by the aircraft, the structural dynamic characteristics of the vehicle in this configuration are altered in such a manner that low flutter speeds can occur within the operational flight envelope. Besides the change in flutter speed, the frequency at which the instability occurs and the aircraft behavior during flutter can be drastically different. Conventional techniques of flutter prevention in general cannot be used. Adding stiffness or mass to the aircraft wing for preventing flutter of one store configuration can adversely affect the dynamic characteristics of other store configurations. Currently, the only suitable means of preventing store flutter is through the use of speed placards. As shown in Figure 1, these speed placards can sometimes be applied deep within the desired operational envelope causing significantly lower aircraft performance for both contemporary and advanced aircraft.

The external store flutter problem is not new. Some contemporary fighter aircraft have flutter restrictions that fall in the range of Mach 0.75 to Mach 1.0; some aircraft/store configurations have flutter limits as low as Mach 0.65. For unusually large, unconventional stores, flutter restrictions occur as low as Mach 0.4. In some cases where classical flutter was not actually encountered, low damping limited the aircraft performance.

Although advanced fighter designs rely on the use of conformal weapon carriage for primary mission effectiveness, it appears that wing

pylon mounted weapons will continue to be used to satisfy secondary missions required for the multirole capability. These aircraft will be designed using very efficient structural optimization techniques for minimum weight and will most probably use composite materials. The wings will be very thin for increased performance, and the engines for these new aircraft will have superior capability. Thus the vehicle will be able to carry external weapons to high speeds without incurring significant drag limitations. This evolution in design procedure will be critical to the external store flutter problem. Metal fighter wings based on strength designs are generally flutter stable (very high "clean wing" flutter margins of safety). However, with the addition of some stores, the flutter margin is completely lost for these metal wings. With the new design capability for reducing aircraft weight and improving performance, flutter and strength are designed concurrently. The flutter speed margin for the clean wing can be designed to be very close to the margins required by military specifications to satisfy the aircraft performance desired. The addition of external stores to this vehicle may be more critical than the metal wing aircraft designed by conventional means. In the past, studies have indicated that flutter speed placards have resulted in speed reductions on the order of 30 percent. In the future, flutter speed restrictions in the transonic range can result in a loss of aircraft speed by as much as 50 percent (Figure 1).

Through the use of active control technology, a concept referred to as active flutter suppression has been shown through analyses and wind tunnel tests to be a promising technique for preventing wing/store flutter problems with minimum aircraft impact (weight, performance, or speed restrictions).

Active Flutter Suppression

Active flutter suppression is a feedback control concept (Figure 2). Transducers embedded in the lifting surface are used to sense disturbances caused by the interaction of the aerodynamic, structural and inertial forces acting upon the vehicle. The signal from the transducer is electronically processed by a computer device. This processing generally includes amplification, phase changes and signal filtering. Various signal filters are used to assure that the active system does not couple with the low frequency flight control dynamics or rigid body modes of the aircraft and does not adversely interact with other aircraft elastic modes. Notch filters may also be required to eliminate local structural resonance problems. Gain and phase networks are required to obtain the proper control surface amplitude and phasing. The compensated signal commands the control surface to oscillate at the proper frequency, creating aerodynamic loads which interact with the lifting surface airloads, thereby preventing flutter.

In addition to feedback compensation, the types and locations of the transducers, and control surface size and location are important design considerations. In most cases, an existing control surface is

used in the design of the system to minimize the impact on vehicle modification. The design technique then becomes the choice of feedback signals and the electronic compensation required for optimum performance for the flight conditions anticipated. For external store flutter suppression, the design of the feedback compensation can become difficult since the compensation may require significant changes for each store configuration. This is due to the external store flutter problem of varying aircraft dynamic characteristics for the different configurations. It is more efficient in terms of cost and performance to change the software of a digital computer or insert an analog patch board for each feedback compensation, than it is to modify the aircraft structure or impose speed placards. Although the compensation changes made in the analyses and during the wind tunnel tests were accomplished manually, the long range goal is to develop an automatic adaptive device that can sense aircraft characteristics and varying flight conditions for making a logical decision for compensation changes.

Background

In 1971, the Air Force Flight Dynamics Laboratory (AFFDL) sponsored a feasibility study to evaluate the potential of applying active control technology to the unstable structural modes associated with fighter aircraft (Reference 1). The results of this program indicated that the milder flutter modes associated with the wing/external store configurations could be prevented very efficiently using active flutter suppression. A follow-on effort focused on a preliminary design (Reference 2) of a wing/store flutter suppression system and provided a comprehensive analytical evaluation of integrating an active flutter suppression system into a representative fighter-attack aircraft.

A joint AFFDL/NASA program reported in Reference 3 improved aeroelastic modeling technology by including miniaturized active controls. A one-thirtieth scale B-52 model was modified to dynamically simulate an aircraft configuration with an active flutter suppression system that was flight tested during the Control Configured Vehicle (CCV) program. The wind tunnel test results showed a significant improvement in damping with the active system operating. A comparison of the measured model and the full scale aircraft damping characteristics showed that the performance of the B-52 CCV airplane flutter suppression system was successfully simulated in the model tests (References 4 and 5).

In 1976, the AFFDL initiated a study which resulted in a wind tunnel demonstration of active wing/store flutter suppression using an aeroelastic model of the YF-17 prototype lightweight fighter. The model had two active control surfaces, one at the leading edge and one at the trailing edge. Three different store configurations were tested in the NASA Langley Research Center 16-Foot Transonic Dynamics Tunnel facility with either the leading edge or trailing edge control surface active. The tests demonstrated that a leading edge control surface can suppress flutter without the aid of a trailing edge surface. The results of this test program are documented in References 6 and 7.

The AFFDL is currently sponsoring four efforts in the area of active flutter suppression. One effort involves a feasibility study for the F-16. A second effort investigates the integration of active flutter suppression and aeroelastic tailoring concepts. The third effort investigates adaptive control concepts that could be used for active flutter suppression. The fourth effort involves additional testing of the YF-17 flutter suppression model to investigate improved control laws, new store configurations, and adaptive control concepts.

NASA has sponsored several efforts in the area of active flutter suppression based on the aerodynamic energy concept proposed by E. Nissim (Reference 8). A wind tunnel model test of this concept was conducted by NASA using a proposed supersonic transport (SST) wing design (Reference 9).

There is also considerable activity in the area of active flutter suppression outside the United States. Reference 10 presents a United Kingdom study on flutter suppression. Significant contributions in terms of wind tunnel test and flight demonstrations by the French and Germans are documented in References 11 and 12. A joint program between the U.S. Air Force and the Federal Republic of Germany Ministry of Defense has been initiated to flight test an active flutter suppression system on an F-4 aircraft for one store configuration. The goal of this program is to demonstrate a flutter speed improvement of 10 percent using the active flutter suppression system.

ANALYSIS PROCEDURE

The automated flutter analysis module (AFAM) of the FASTOP computer program (Reference 13) was used to perform the unaugmented flutter calculations reported herein and to provide unsteady aerodynamics for the frequency domain evaluations of the active flutter suppression system. The subsonic doublet lattice option was used to calculate the unsteady aerodynamics. For the rigid body, elastic, and control surface modes required for the analysis, unsteady aerodynamic matrices for an assumed set of reduced velocities (which cover the frequency and velocity ranges of interest) are calculated and stored on a data tape for later use. An additional 200 sets of aerodynamics are interpolated using the original set of data for the frequency domain studies. The analysis procedure is shown as a schematic in Figure 3. If measured wind tunnel stability derivatives are available, a doublet lattice aerodynamic program referred to as H7WC (Reference 14) can be used to tune the aerodynamic representation (box patterns) before employing the FASTOP program for the flutter analysis.

For the frequency domain analyses, the control system interaction program, ACF (Reference 15), developed for the earlier active flutter suppression studies (Reference 1) sponsored by the AFFDL was used.

To determine the augmented aircraft stability characteristics, modified Nyquist criterion (Reference 16) is used. From Figure 4, the closed loop transfer function (output/input) of the system is defined as

$$\frac{C(S)}{R(S)} = \frac{G(S)}{1+G(S)H(S)}, \tag{1}$$

where G(S) represents a product of all control system transfer functions in the forward loop, and H(S) represents the product of all control system transfer functions in the feedback loop. The aircraft dynamics are normally included in G(S).

The denominator of equation 1 can be defined as F(S) and expressed as

$$F(S) = \frac{(S+Z_1)(S+Z_2)\cdots(S+Z_m)}{S^J(S+P_1)(S+P_2)\cdots(S+P_n)} . \tag{2}$$

When F(S) is set equal to zero, the resulting expression is called the characteristic equation of the system. The zeros of F(S) are defined as the roots of the characteristic equation and can be real or complex numbers. For a stable closed loop system, all the roots (zeros) of F(S) must have negative real parts. There is no particular restriction on the poles of F(S). However, the poles of F(S) are also the poles of G(S)H(S) which is defined as the open loop system. If any of the poles of G(S)H(S) lie in the right half of the S-plane (positive real parts) the open loop system will be unstable. Nevertheless, the closed loop system can still be stable if all the zeros of F(S) are found in the left half of the S-plane.

The objective of modified Nyquist theory is to determine if any of the zeros of the characteristic equation (F(S) = 0) lie in the right half of the S-plane. The approach is to define a trace (Nyquist path) which encloses the entire right half of the S-plane without passing through any of the poles or zeros of F(S) which may lie on the imaginary axis or at the origin. Once the Nyquist path is specified, the stability of the closed loop system can be determined by plotting F(S) as S takes on values along the Nyquist path, and investigating the behavior of F(S) plot with respect to the origin of the F-plane. The F(S) locus mapped into the F-plane will encircle the origin as many times as the difference between the number of zeros and the number of poles of F(S) that were encircled by the Nyquist path. In other words,

$$N = Z - P \tag{3}$$

where N = the number of encirclements of the origin made by the F(S) locus in the F-plane

Z = number of zeros of F(S) on the right-hand side of the S-plane

and P = number of poles of F(S) on the right-hand side of the S-plane.

For a stable system, Z must always be zero which implies that the Nyquist plot of F(S) must encircle the origin P times. The encirclements, if any, must be made in a negative (counter clockwise, N = - P) direction, since the Nyquist path in the S-plane was assumed in the clockwise direction. If the assumed aircraft speed falls below flutter onset, there are no poles in the function G(S) that lie in the right half of the S-plane. Generally, there are also no poles of H(S) in the right half of the S-plane as can be determined from inspection of the feedback stability augmentation system. For this case, P = 0. As a result, any clockwise encirclement of the origin indicates the existence of a zero in the right half of the S-plane, and therefore an unstable condition. If the assumed aircraft speed is above the passive flutter speed, there will be pole(s) in the right half of the S-plane; in this case, there must be a counter clockwise encirclement of the origin for each pole for the system to be stable. The frequency of the critical mode can be found by locating the point at which F(S) crosses the real axis of the Nyquist plot (the value of frequency required to cause F(S) to be real).

APPLICATION

AFFDL Wind Tunnel Model

Under AFFDL sponsorship, Northrop Corporation designed a wind tunnel model (Figure 5) and the active control laws for a wind tunnel test program to investigate wing/store active flutter suppression (Reference 7). The model, fabricated by Dynamics Technology Inc., is a semispan representation of a lightweight fighter (a 30% length scale YF-17 with the exception of the specially designed control surfaces and the addition of an outboard store pylon location). Partial span leading edge and trailing edge control surfaces were used as the active flutter suppression surfaces. These control surfaces were driven by miniature hydraulic actuators (fabricated by The Boeing Wichita Company) located near the hingelines. A non-dynamically scaled horizontal tail was used to trim the flutter model at various test conditions. The model wing had an aluminum spar with ballasted balsa wood sections attached for shape and mass simulation. The wing tip launcher was flexible; however, the pylons were designed to be relatively rigid. The external stores consisted of a rigid AIM-7S, a rigid AIM-9E and a flexible AIM-9E.

A schematic of the model is presented in Figure 6 and shows overall dimensions. Three different store configurations with different flutter

frequencies were selected for testing to simulate symmetric flutter modes of various degrees of violence (change in damping with respect to a change in velocity at g = 0). These store configurations are described in Table 1.

The control law shown in Figure 7 illustrates the flutter suppression scheme employed in the tests and this study. Accelerometers located on the wing along a streamwise chord just outboard of the control surfaces (Figure 6) were used to sense the wing dynamic response. The difference between these two accelerometers (in g's) was fed back through compensation networks to command the oscillation of either the leading edge or trailing edge control surface. Figure 7 and Table 2 summarize the compensation used for the three different store configurations.

The doublet lattice representation of the wind tunnel model is shown in Figure 8. The model was divided into 17 aerodynamic panels (one for the wing leading edge extension, eight for the wing, one for the tip missile/launcher, two for the horizontal tail, and five for the fuselage interference panels). The panels were further subdivided into aerodynamic boxes. The aerodynamic effects of the 5° wing anhedral and the horizontal tail offset from the fuselage centerline were taken into account. In this analysis, it was assumed that the aerodynamic interference between the store/pylon and the wing was negligible. The tip missile and launcher were represented aerodynamically as flat plates.

Unaugmented flutter analyses (active system off) were conducted for each of the three store configurations at the test Mach numbers using the FASTOP P-k option. For Configuration A, the calculated flutter mode was very mild with a maximum positive damping not exceeding 0.059. For zero structural damping, flutter was predicted at a velocity of 203 KEAS at 12.4 Hz. Configuration B was predicted to flutter in a very violent mode at 150 KEAS at 6.7 Hz for zero structural damping. Flutter was predicted to occur at a speed of 174 KEAS at a frequency of 10.4 Hz for Configuration C.

The active flutter suppression analyses were conducted in the frequency domain using calculated modal characteristics. Figures 9 through 14 present the results (modified Nyquist plots and damping trends) of the flutter suppression analyses and provide comparisons between the calculated and test data for each of the three external store configurations (90 sec movie showing wind tunnel test).

The results for Configuration A are presented in Figures 9 and 10. Figure 9 shows modified Nyquist plots for increasing airspeed. The active surface in this case was the trailing edge control surface. At speeds below the passive flutter condition (approximately 203 KEAS), the Nyquist plot for the critical mode (flutter mode) forms a clockwise loop for increasing frequencies. As the assumed speed approaches the

flutter velocity, the plot grows in response and continues to form a loop in the clockwise direction. At a speed increment slightly above flutter, the Nyquist plot has a counter clockwise encirclement of the origin indicating a stable condition with the system operating. Analyses indicate that this configuration would be stable to speeds significantly higher than 240 KEAS with the system operating; the improvement in flutter speed would be greater than 18%. Figure 10 shows the experimental results with the trailing edge flutter suppression system on and off. During the wind tunnel tests, Configuration A did not flutter due to the unexpectedly high damping attributed to the model support system. It can be seen from the test data, however, that the trailing edge system added significant damping to the critical mode.

Figures 11 and 12 show similar calculated and test results for Configuration B for a Mach number of 0.8. Figure 11 shows the Nyquist plot variation with airspeed for an active leading edge control system. For this configuration, flutter was predicted to occur at approximately 150 KEAS. The critical loop of the Nyquist plot tended to rotate in a clockwise direction about the (+1,0) point as the velocity was increased. With additional phase lag (30° at approximately 6 Hz) in the control laws, the augmented flutter speed could be further increased. However, the critical loop becomes quite narrow, which would indicate very low phase margins. For the leading edge control law analyzed, the flutter speed improvement was predicted to be about 19%. Figure 12 shows test data plotted versus airspeed for the system both on and off. From the test data, the projected augmented flutter speed was 171 KEAS. Therefore, the test data showed a flutter speed improvement of approximately 9%. The predicted system-on instability (179 KEAS) was 4.7% higher than the similar condition projected from test data (171 KEAS).

The analysis and test data for Configuration C at a Mach number of 0.6 are presented in Figures 13 and 14. Figure 13 presents the Nyquist plot variation with airspeed for a leading edge control system. Flutter, in this case, was predicted to occur at a speed of 174 KEAS. It is interesting to note that the Nyquist plot rotated in a clockwise direction about the (+1,0) point with increasing airspeed. It would have been beneficial in this case, in terms of increased flutter speed, if about 15° phase lead at 10 Hz were added to the control law. Once again the model would have been stable for speeds exceeding 225 KEAS but with very low gain and phase margins. Figure 14 presents the test results for Configuration C with the active leading edge control law. The projected system-off and system-on instabilities from test data were 176 KEAS and 193 KEAS, respectively. This was about a 10% increase in flutter speed. The analyses predicted passive flutter at 174 KEAS and augmented flutter at 219 KEAS for an increase in flutter speed of 26%.

In general, control surface aerodynamic force and moment coefficients predicted by theory are high when compared to experimental data. Unmodified control surface aerodynamics, therefore, tend to predict a more efficient active flutter suppression system. Since experimental control

surface aerodynamics were not available for this model, there was no attempt in this study to reduce the magnitude of the control surface aerodynamics. This may be a partial explanation of the differences obtained between the analysis and test results with the active flutter suppression system on.

CONCLUDING REMARKS

This paper presented the results of an analytical study of a wing/store active flutter suppression wind tunnel model. Three different external store configurations were analyzed. Leading edge and trailing edge control surfaces, each acting independently, were used in the analysis. The control laws which were found to be most effective during the wind tunnel tests were analyzed; the resulting calculated speed improvements were compared with available test data. In all cases, the predicted passive flutter speed for each store configuration using calculated mode characteristics was conservative when compared to test data. Calculations with the active flutter suppression system operating indicated a larger flutter speed improvement than the projected test data showed. This was expected because of the use of unmodified control surface aerodynamics in the analysis.

This paper has shown that Nyquist stability criteria is a valuable tool for the design and analysis of active flutter suppression systems. However, care must be taken to assure that the aerodynamics, the structural dynamics and the control system representations of the aircraft are well defined through comparisons with available experimental data.

The Air Force has plans to continue testing the flutter model described herein to investigate improved single surface control laws and control laws which use both the leading edge and trailing edge control surfaces acting together. The most challenging aspect of wing/store flutter suppression appears to be automatic adaptive control. Plans are to further investigate adaptive control devices with the objective of implementing the most promising technique for demonstration in the wind tunnel. Also, additional work is required to improve and evaluate unsteady aerodynamic prediction methods for oscillating control surfaces.

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CONTEMPORARY AND ADVANCED AIRCRAFT ENVELOPES
WITH WING MOUNTED EXTERNAL STORES

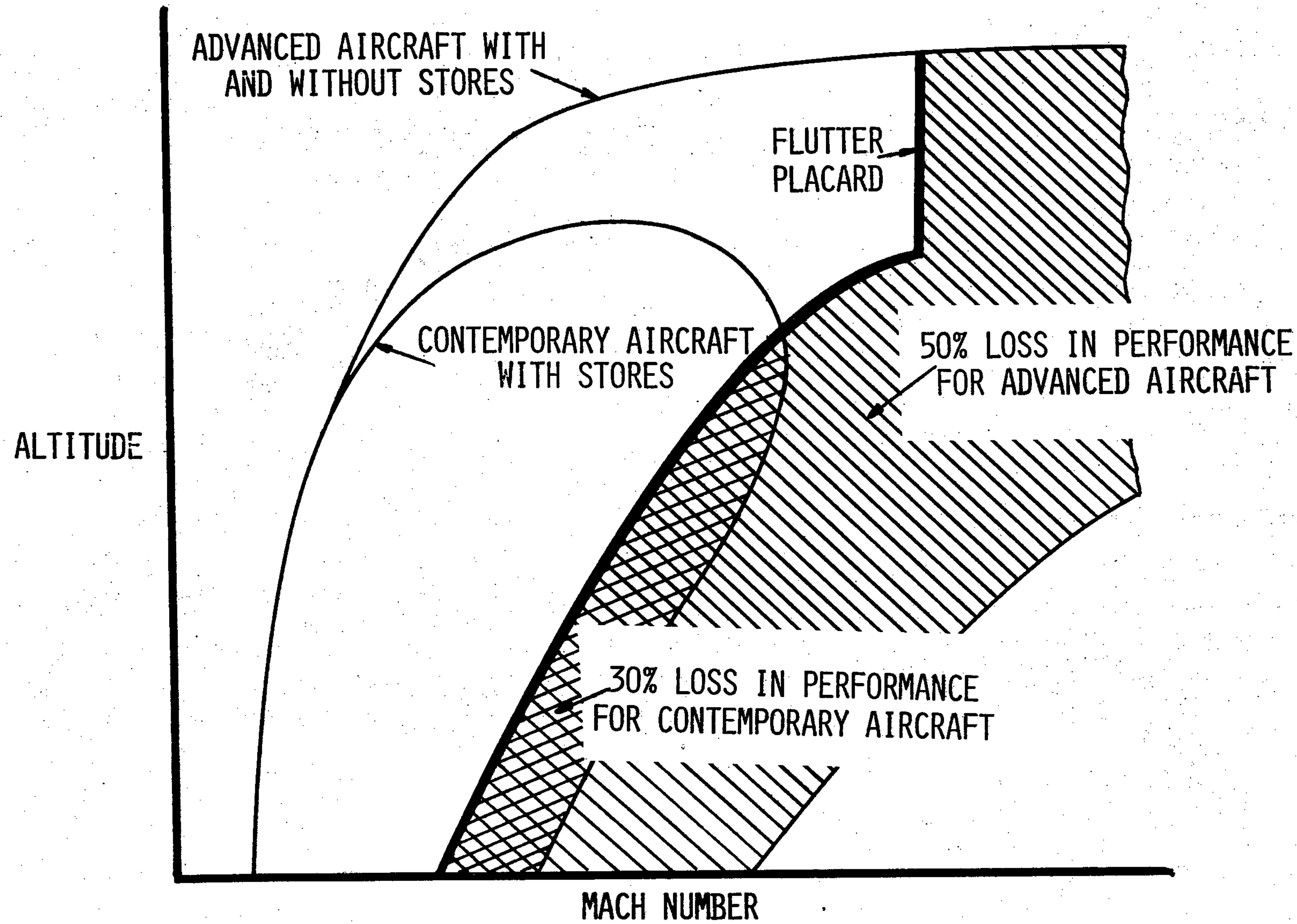


Figure 1

CONCEPTUAL DIAGRAM OF ACTIVE FLUTTER SUPPRESSION

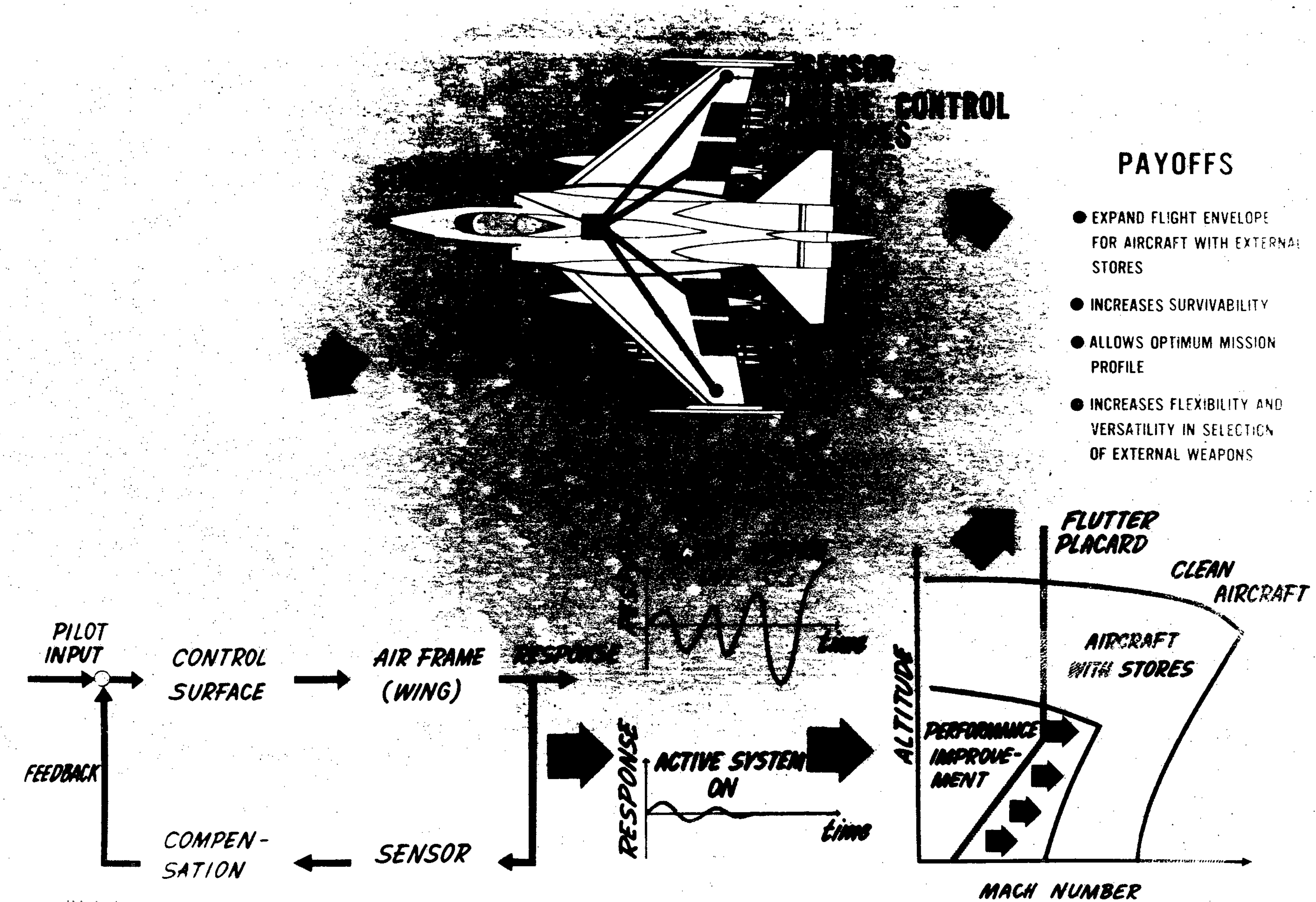


Figure 2

AEROSERVOELASTIC ANALYSIS PROCEDURE

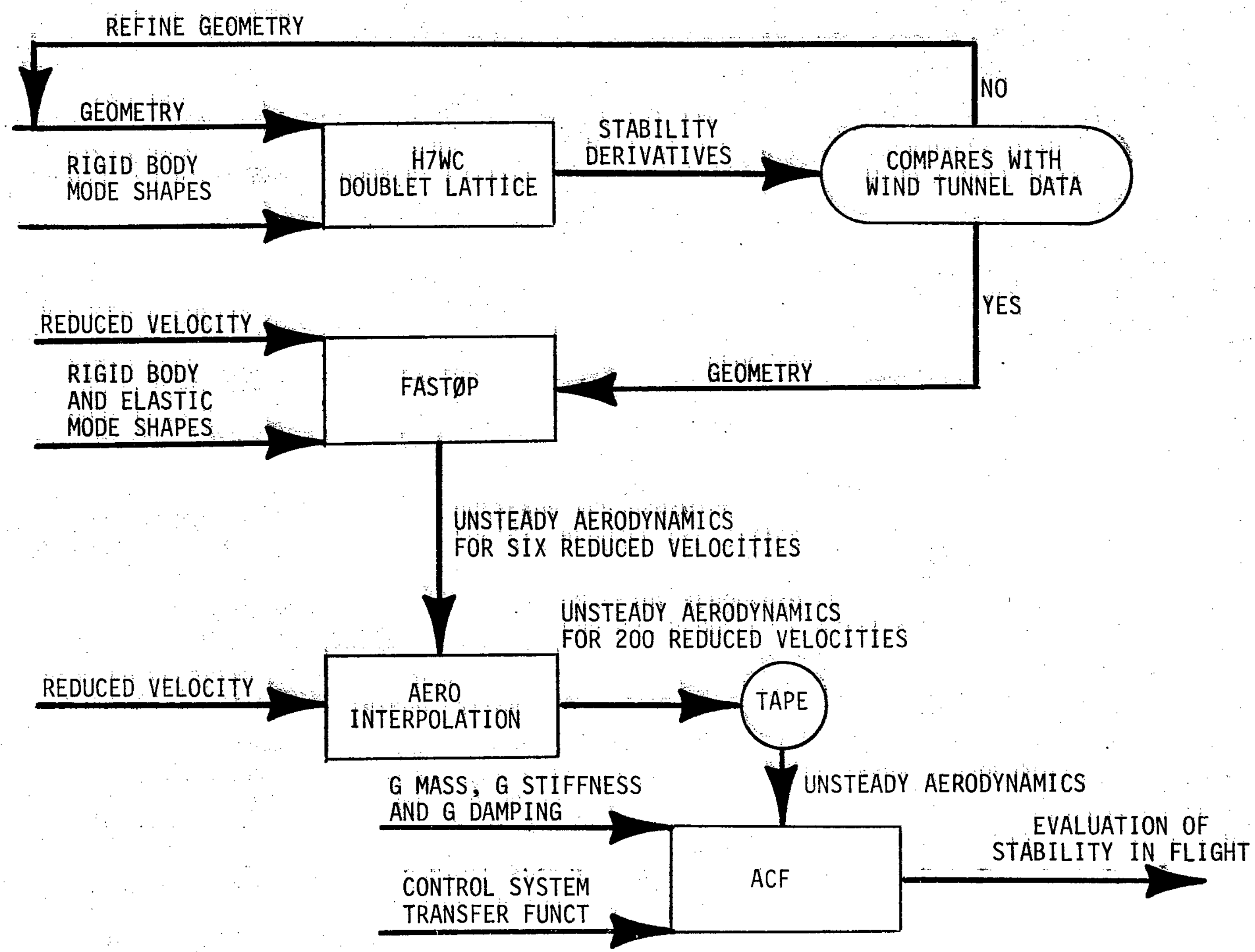


Figure 3

MODIFIED NYQUIST CRITERIA

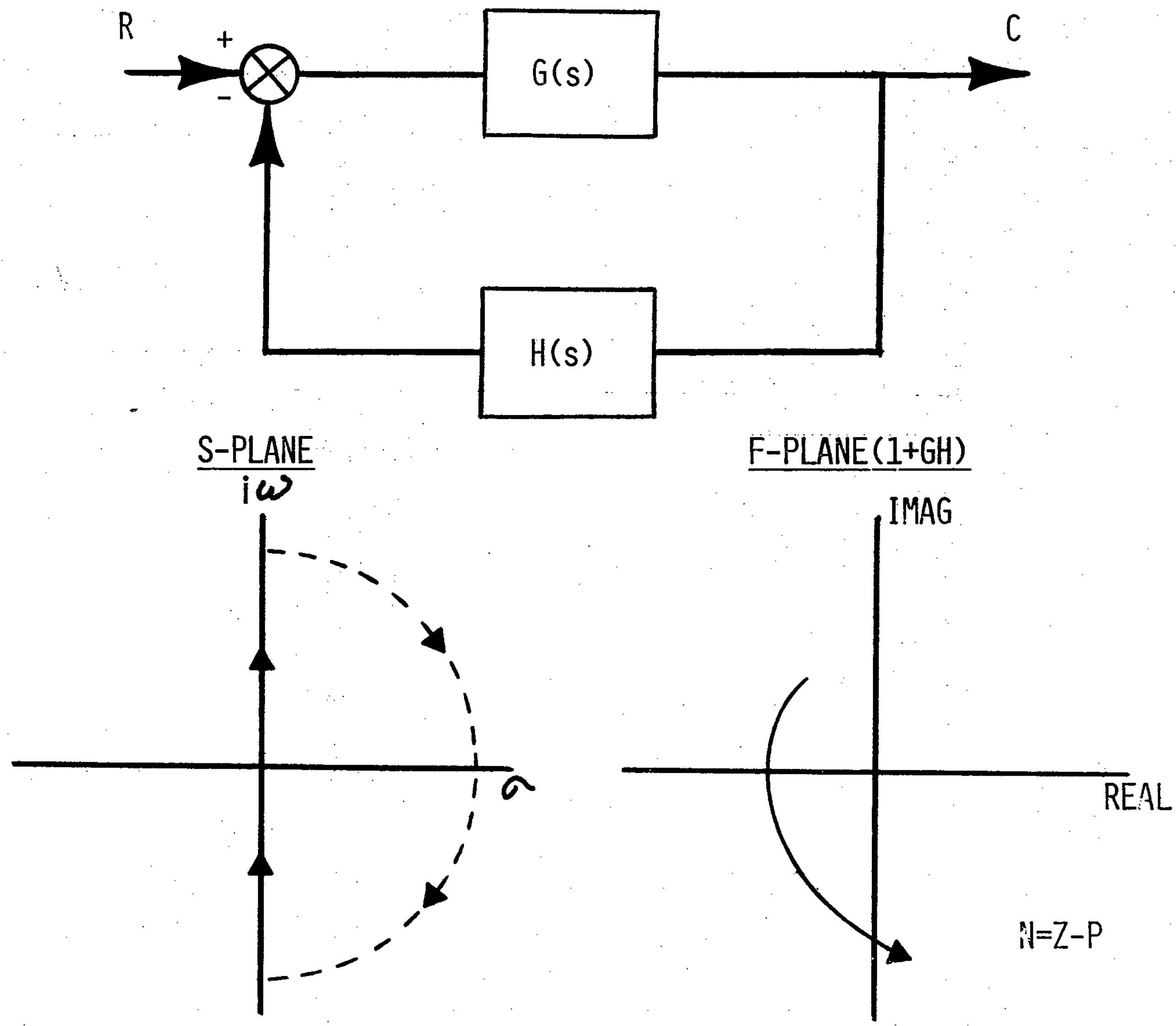


Figure 4

YF-17 WIND TUNNEL MODEL

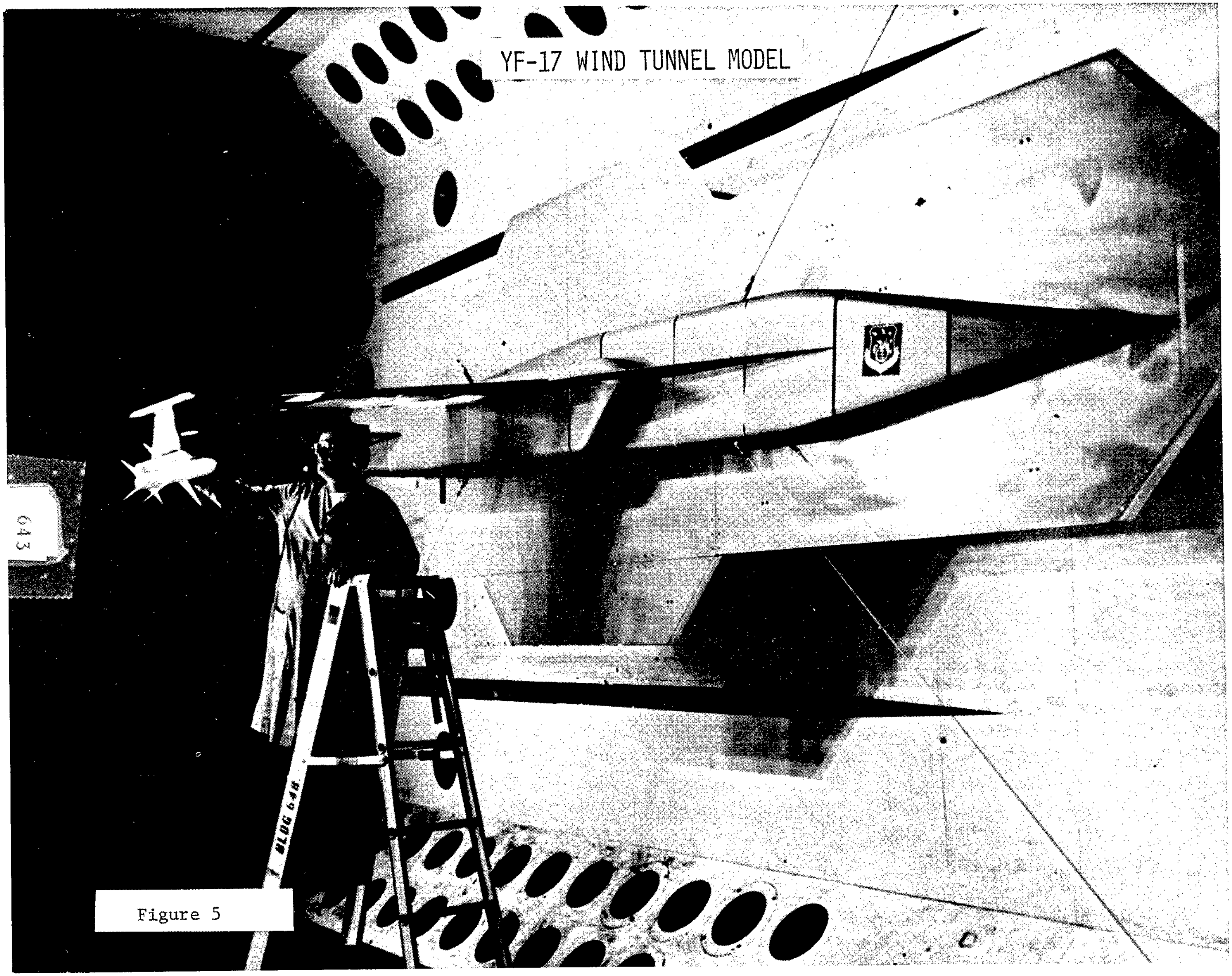
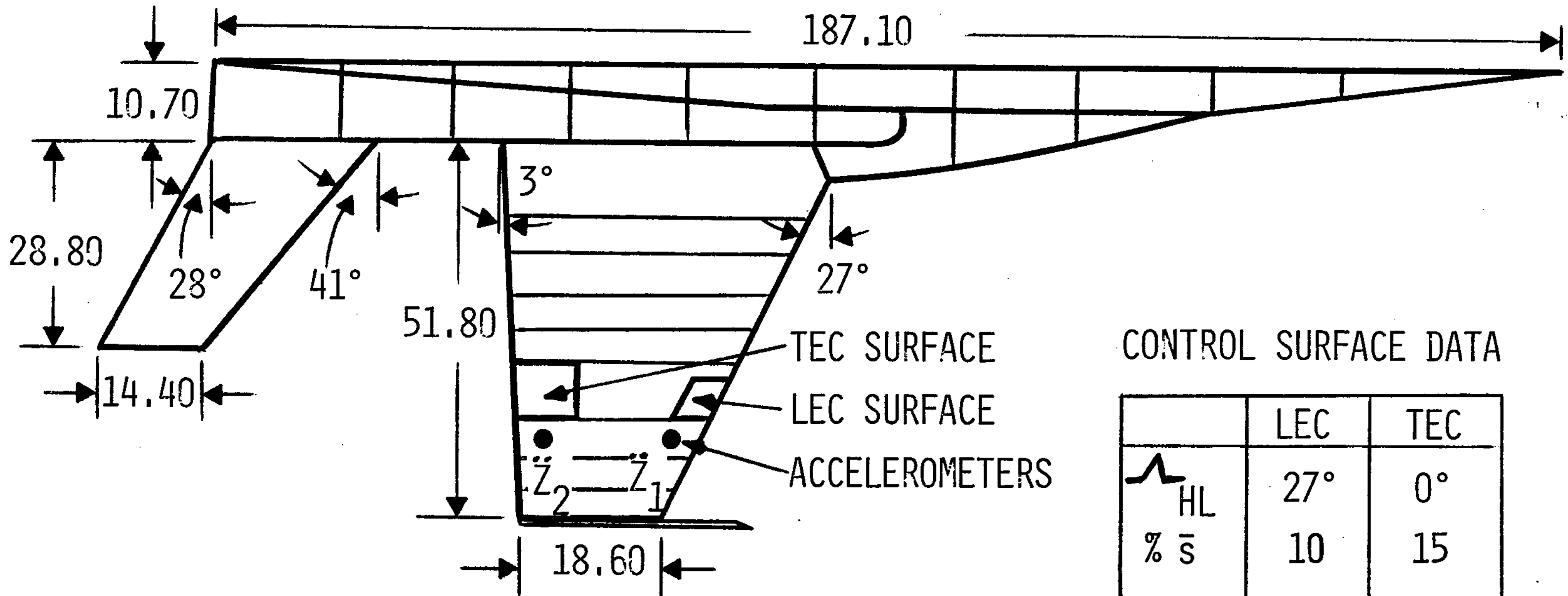


Figure 5

WIND TUNNEL MODEL OVERALL PLANFORM DIMENSIONS



ALL DIMENSIONS IN INCHES

CONTROL SURFACE DATA

	LEC	TEC
Λ_{HL}	27°	0°
$\% \bar{s}$	10	15
$\% \bar{c}$	20	30

Figure 6

ACTIVE FLUTTER SUPPRESSION BLOCK DIAGRAM

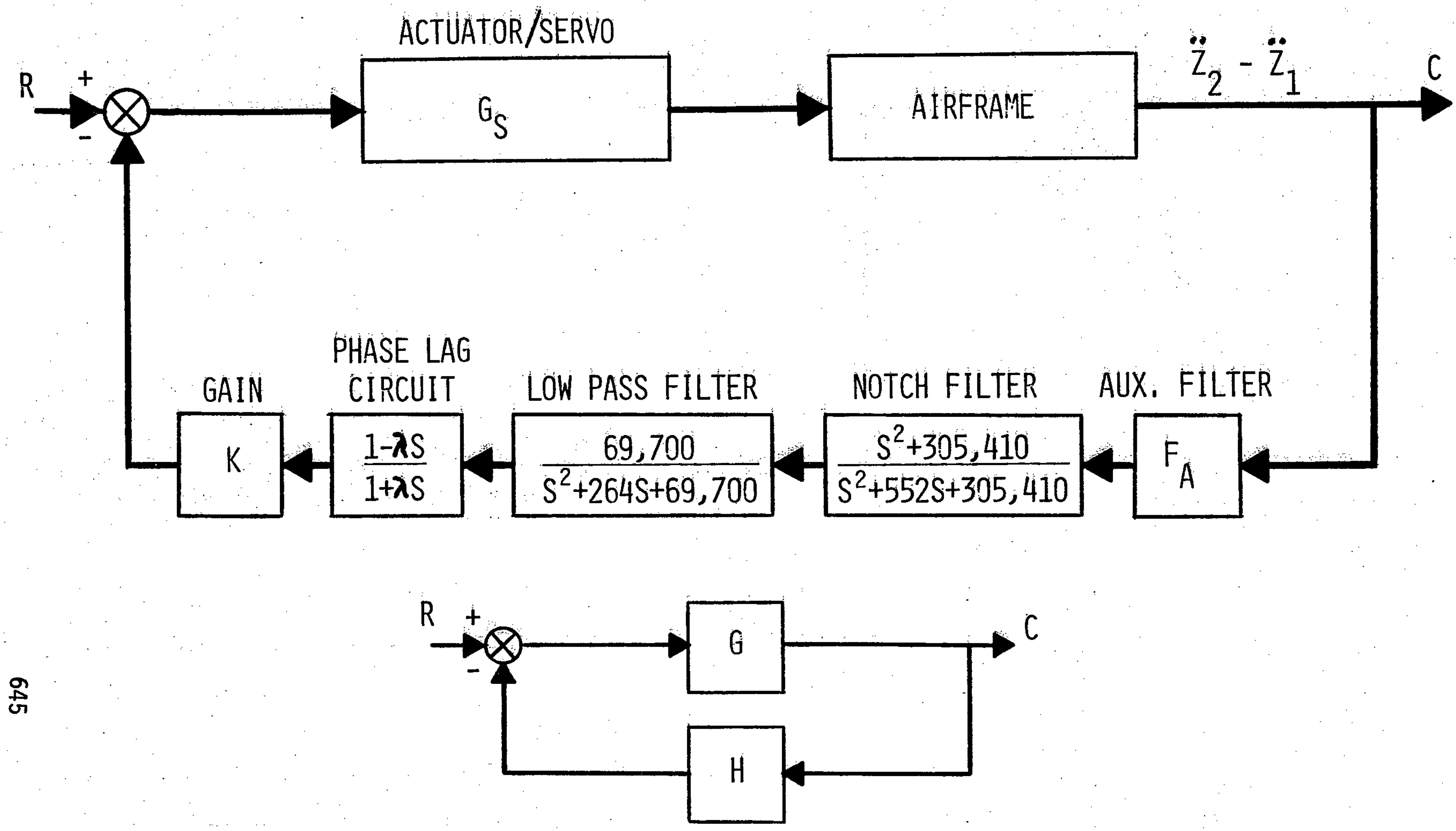


Figure 7

AERODYNAMIC REPRESENTATION OF THE FLUTTER MODEL

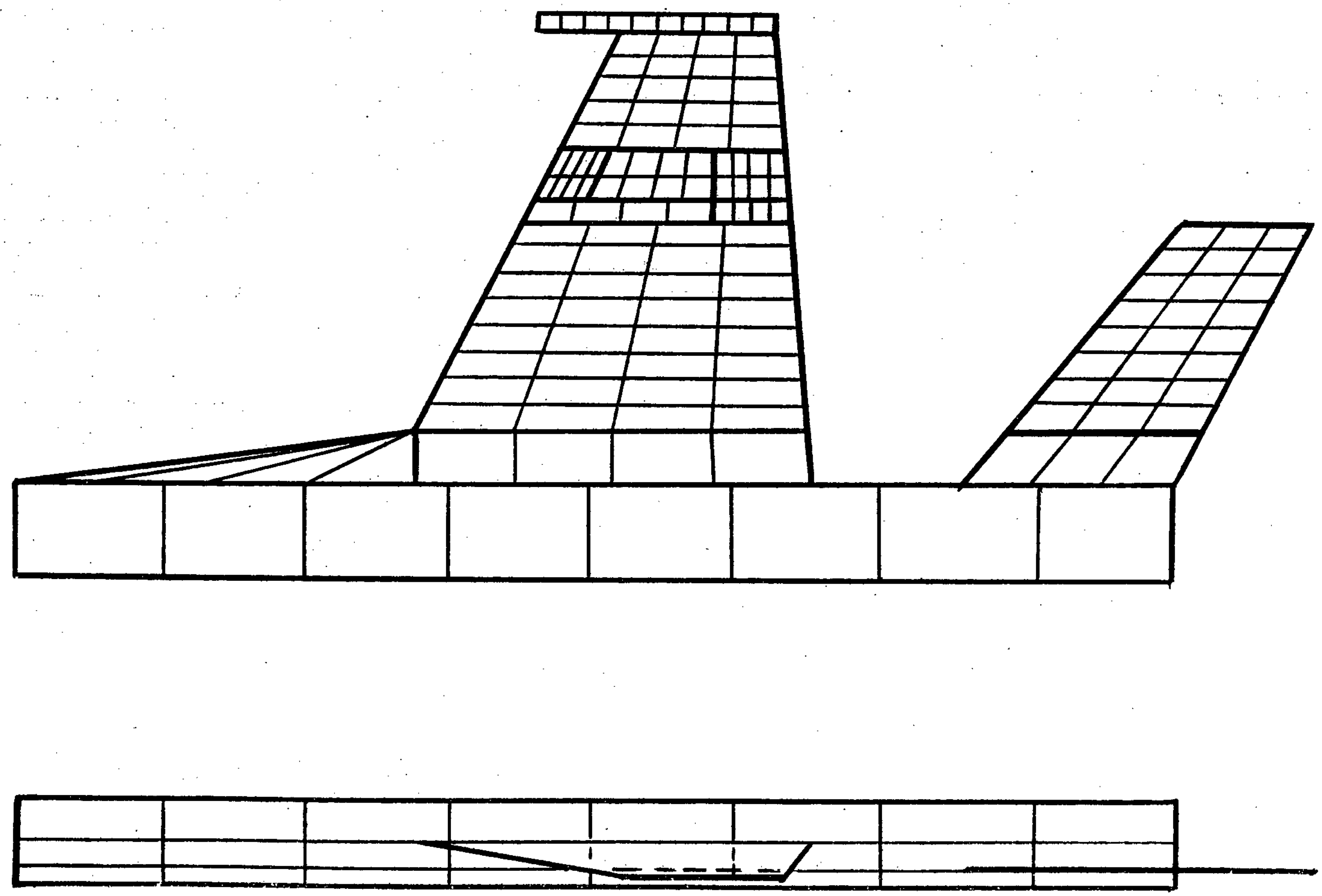


Figure 8

MODIFIED NYQUIST PLOTS FOR CONFIGURATION A,
TRAILING EDGE CONTROL LAW, M = 0.8 (SEA LEVEL)

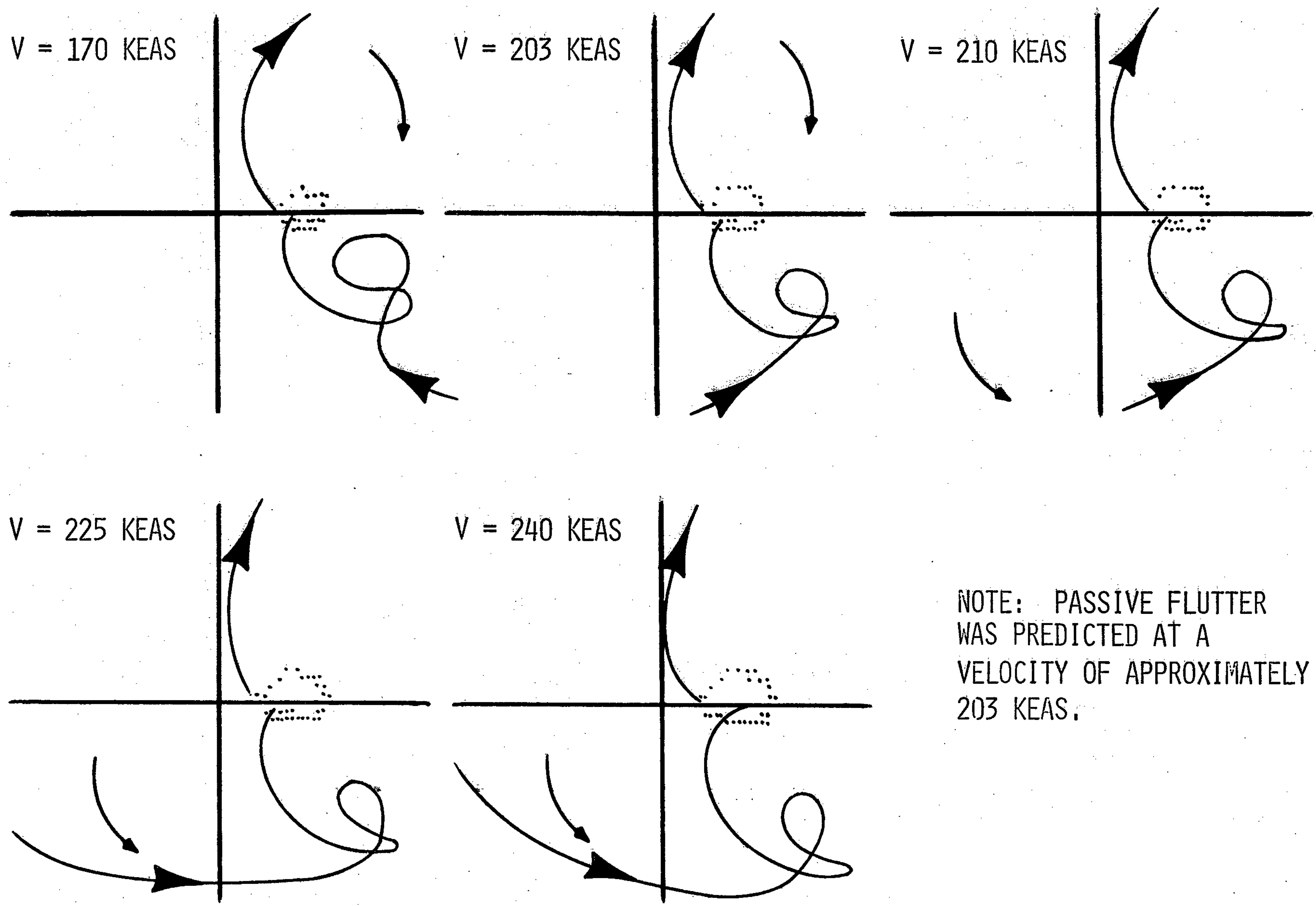


Figure 9

COMPARISON OF CALCULATED AND TEST RESULTS FOR CONFIGURATION A, M = 0.8

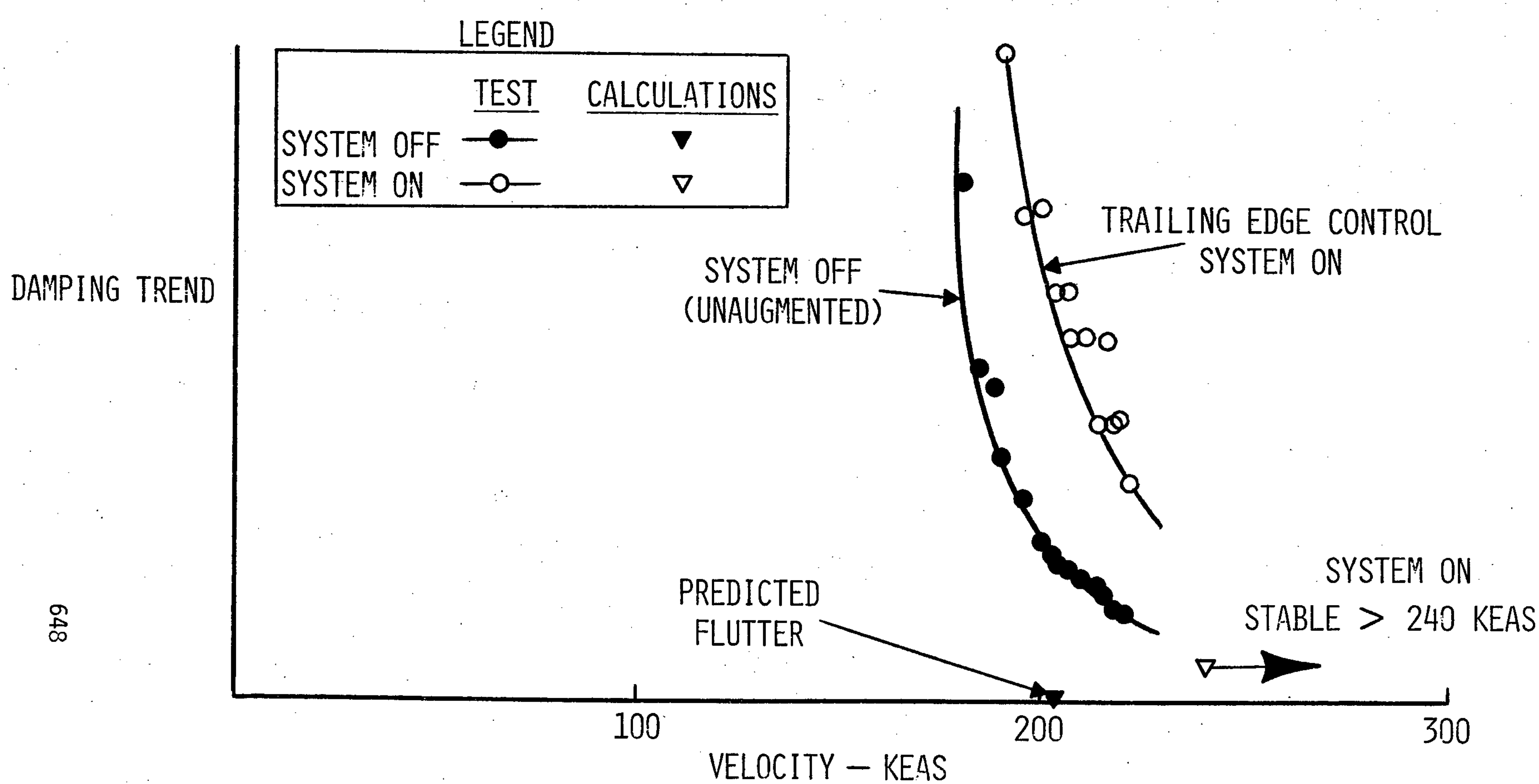
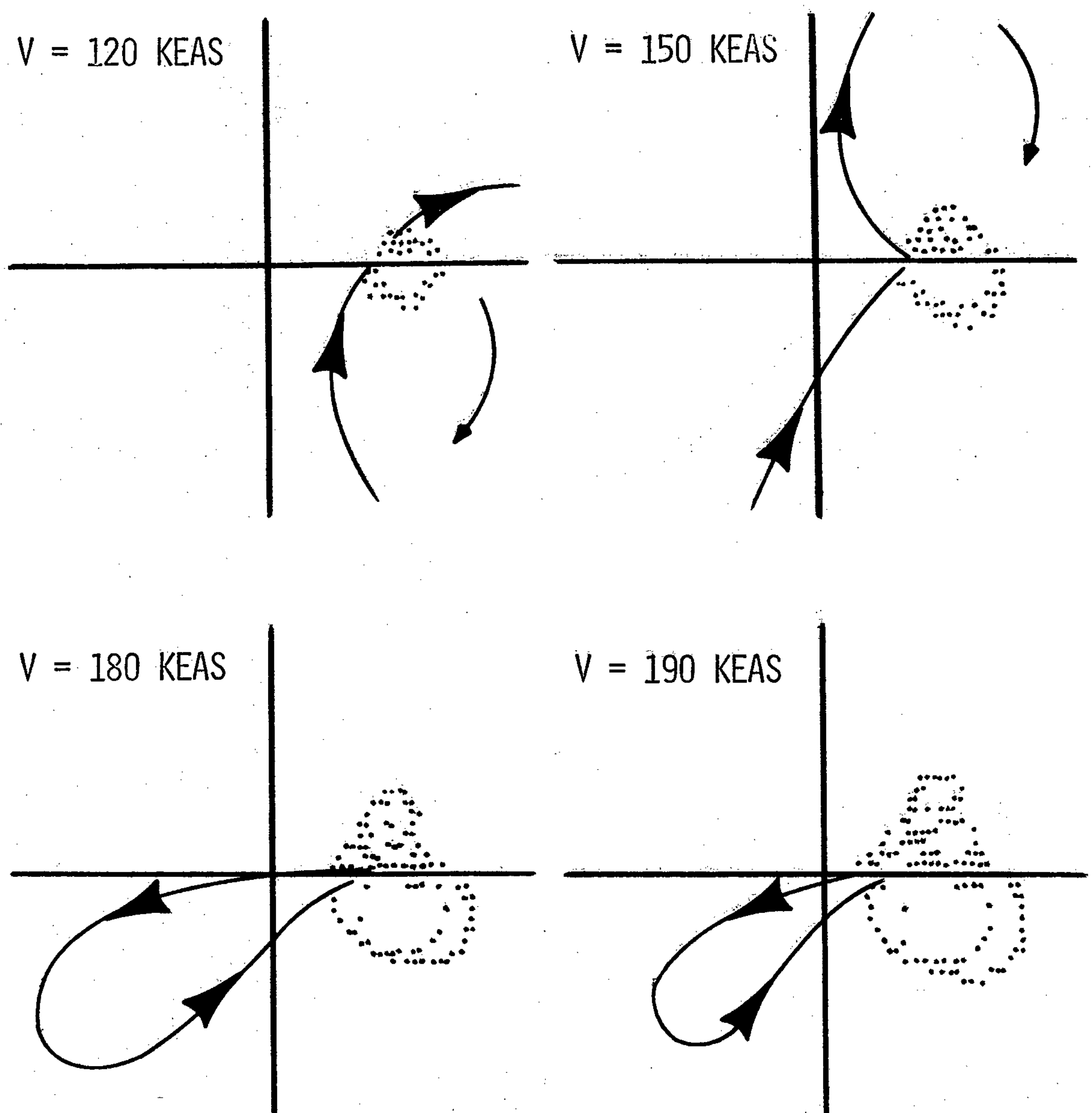


Figure 10

MODIFIED NYQUIST PLOTS FOR CONFIGURATION B,
LEADING EDGE CONTROL LAW, $M = 0.8$ (SEA LEVEL)



NOTE: PASSIVE FLUTTER WAS PREDICTED AT A VELOCITY OF APPROXIMATELY 150 KEAS.

Figure 11

COMPARISON OF CALCULATED AND TEST RESULTS FOR CONFIGURATION B, M = 0.8

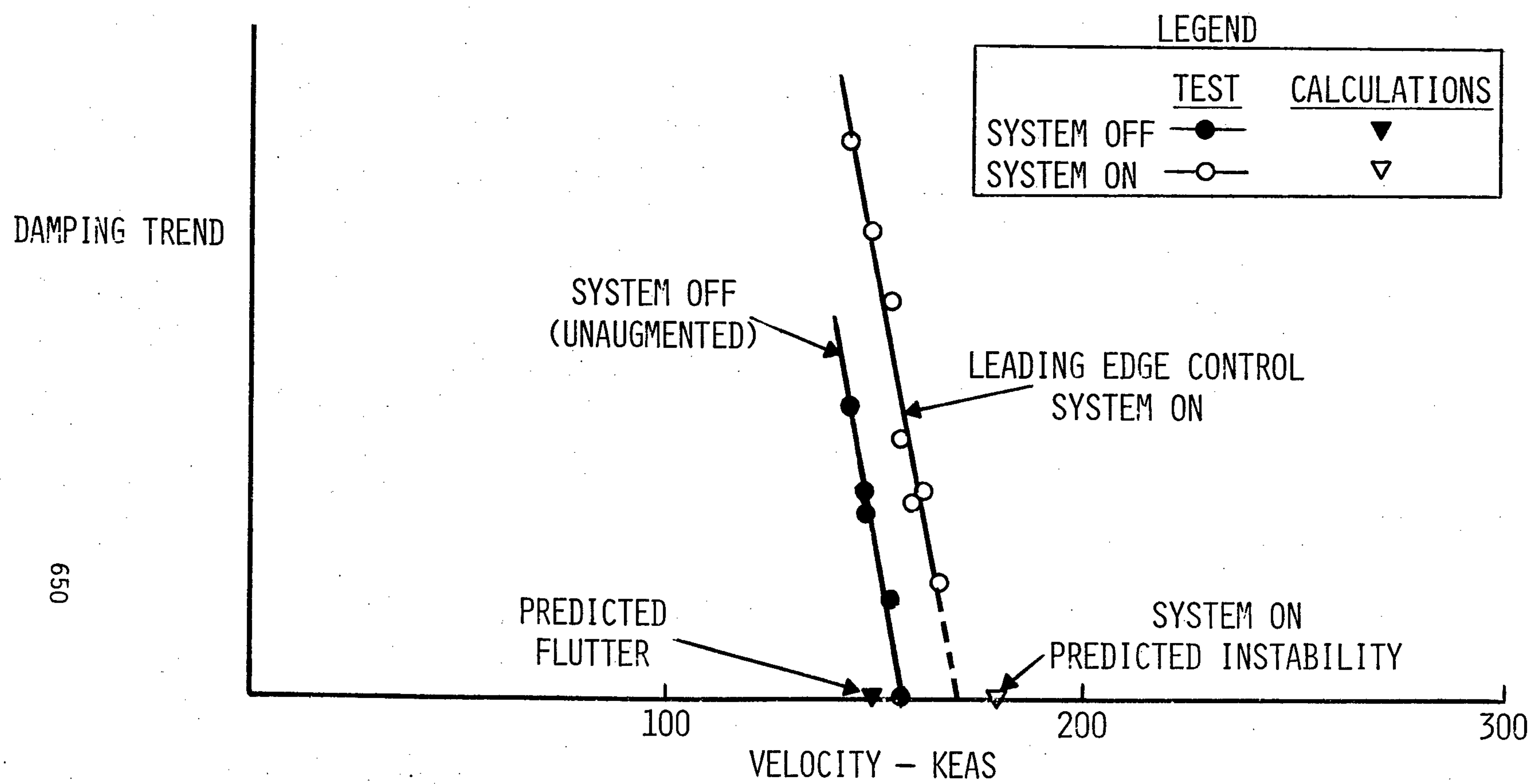
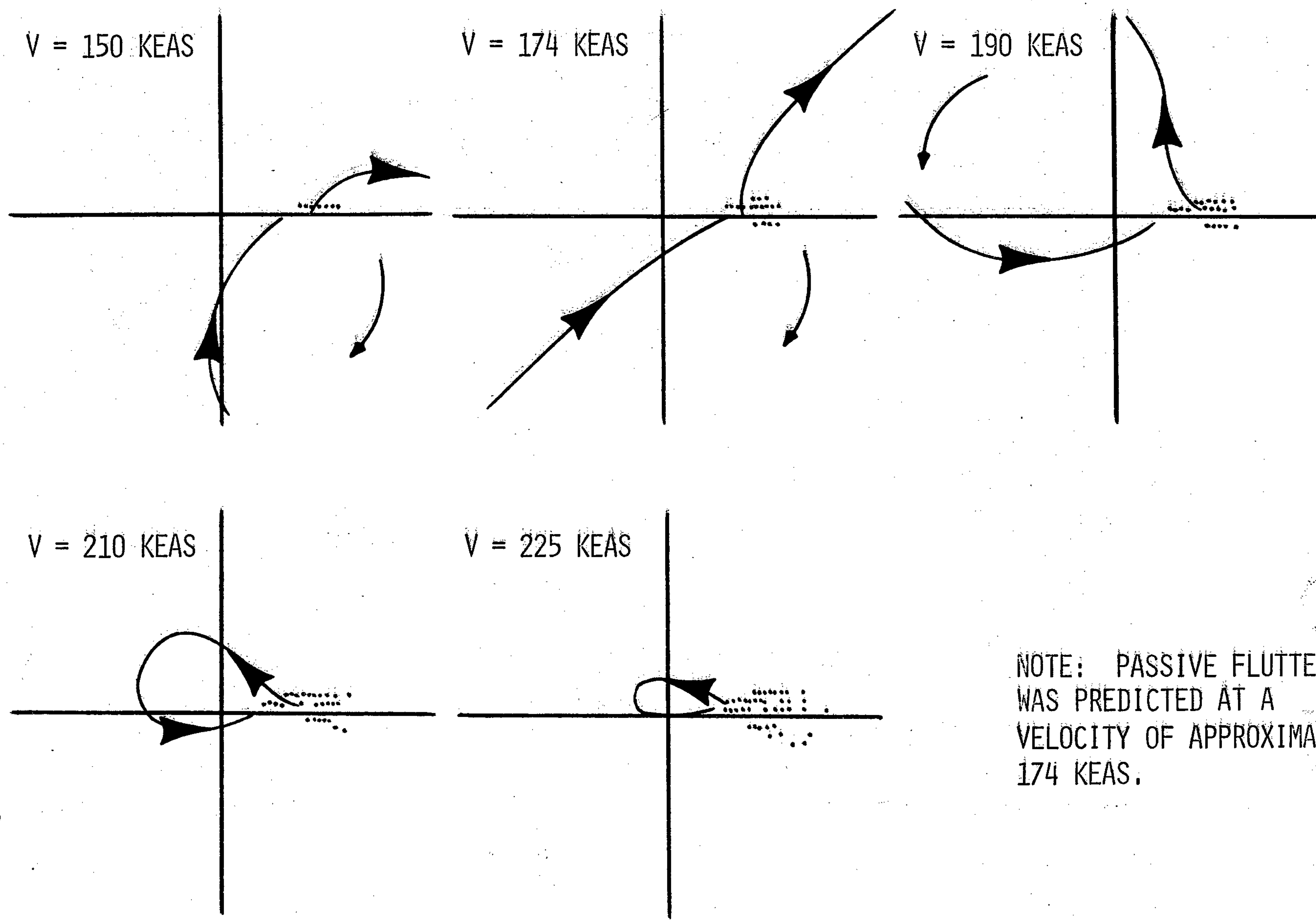


Figure 12

MODIFIED NYQUIST PLOTS FOR CONFIGURATION C, LEADING EDGE CONTROL LAW, M = 0.6 (SEA LEVEL)



NOTE: PASSIVE FLUTTER WAS PREDICTED AT A VELOCITY OF APPROXIMATELY 174 KEAS.

Figure 13

COMPARISON OF CALCULATED AND TEST RESULTS FOR CONFIGURATION C, M = 0.6

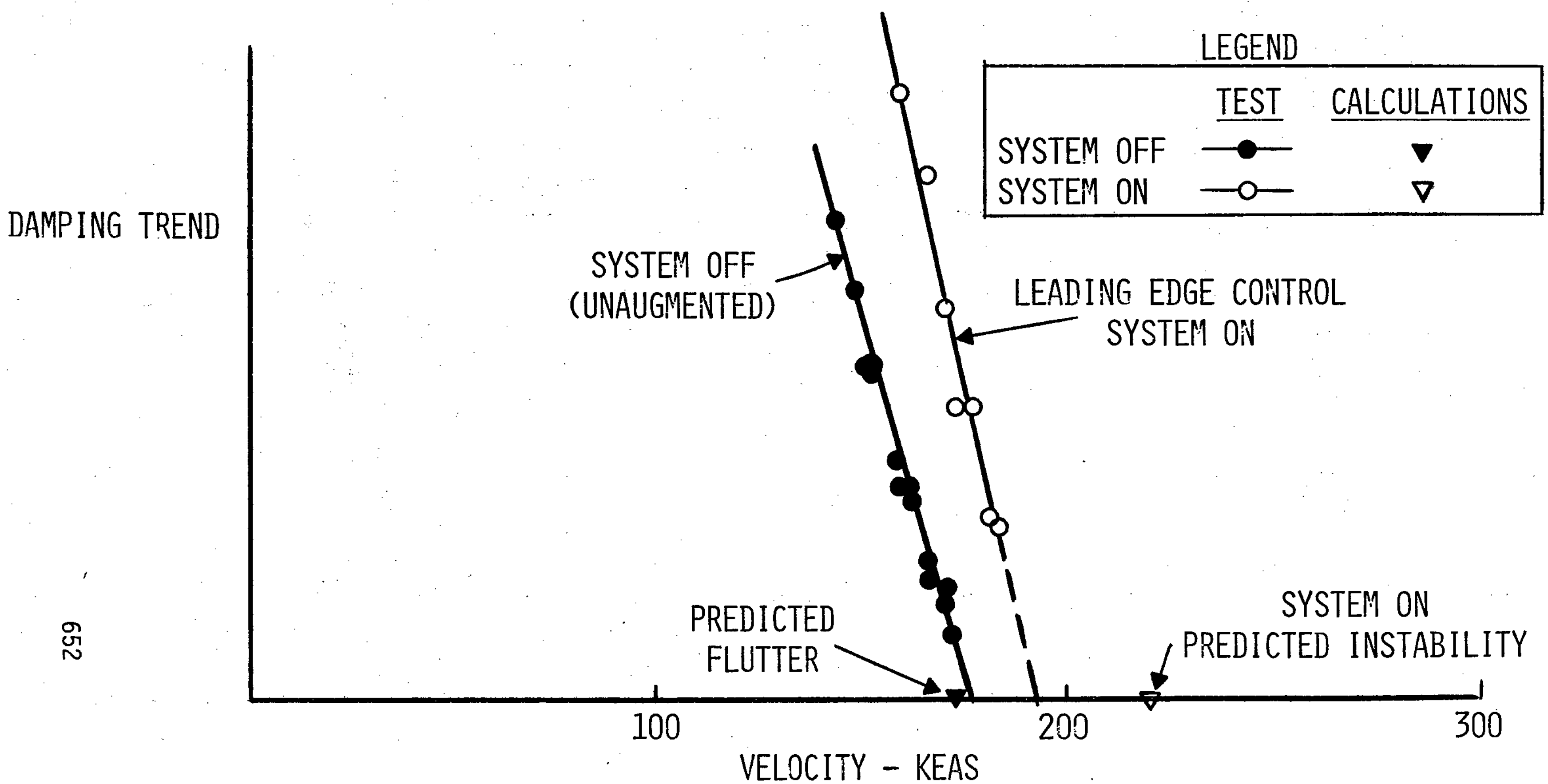


Figure 14

DESCRIPTION OF TEST CONFIGURATIONS

CONFIGURATION A

TIP LAUNCHER RAIL:	AIM-9E (FLEXIBLE)
INBOARD PYLON:	AIM-7S (RIGID)
OUTBOARD PYLON:	NOT INSTALLED

CONFIGURATION B

TIP LAUNCHER RAIL:	EMPTY
INBOARD PYLON:	NOT INSTALLED
OUTBOARD PYLON:	AIM-7S (RIGID)

CONFIGURATION C

TIP LAUNCHER RAIL:	EMPTY
INBOARD PYLON:	NOT INSTALLED
OUTBOARD PYLON:	AIM-9E (RIGID)

Table 1

BLOCK DIAGRAM TRANSFER FUNCTIONS FOR EACH STORE CONFIGURATION

ELEMENT	CONF A	CONF B	CONF C
F_A	$\frac{0.1S}{1+0.02S}$	$\frac{S}{(1+0.03S)(1+0.015S)}$	$\frac{S}{(1+0.02S)(1+0.012S)}$
λ	0.0439	0.0274	0.0108
K	0.760	0.532	0.190

$$G_S = \begin{cases} \dfrac{3,097,600 \; G_N}{(S+100)(S^2+88S+30,976)} & \text{FOR LEADING EDGE SYSTEM} \\[2ex] \dfrac{1,790,278 \; (S+36) \; G_N}{(S+125)(S+33)(S^2+125S+15,600)} & \text{FOR TRAILING EDGE SYSTEM} \end{cases}$$

$$G_N = \frac{S^2+21S+45,590}{S^2+299S+45,590}$$

Table 2

Biographical Sketch

Mr. Lawrence J. Huttsell was born in Ansley, Nebraska on November 19, 1944. He graduated from the University of Louisville in 1967 with a Bachelors Degree in Mechanical Engineering. In 1972, he received a Masters Degree in Aeronautical and Astronautical Engineering from Ohio State University.

In June 1967, Mr. Huttsell accepted a position at the Air Force Flight Dynamics Laboratory. He has been involved with programs in flutter, subsonic and supersonic unsteady aerodynamics, and aeroservoelasticity. His recent work has involved analysis and test of wing/store active flutter suppression concepts. He is a member of the American Institute of Aeronautics and Astronautics and the American Society of Mechanical Engineers.

Mr. Thomas E. Noll was born in Cincinnati, Ohio on August 31, 1944. He graduated from the University of Cincinnati receiving a B.S. degree in Mechanical Engineering in 1967. Mr. Noll accepted a position at the Air Force Flight Dynamics Laboratory soon after graduation. In 1972, Mr. Noll received his M.S. degree in Aeronautical and Astronautical Engineering from Ohio State University.

In 1971, Mr. Noll was assigned project engineer in the technical area of active flutter suppression. He has published twelve reports or papers in this area and related fields since 1972. Mr. Noll is currently the USAF project engineer for a joint U.S./Federal Republic of Germany research program to demonstrate active flutter suppression through flight tests on an F-4 aircraft. He is also responsible for the joint AFFDL/NASA wind tunnel programs for evaluating active flutter suppression concepts on a lightweight fighter configuration. Mr. Noll is a member of the American Institute of Aeronautics and Astronautics, and a member of the Structural Dynamics Technical Committee of that society.

Biographical Sketch

Mr. Cooley is Technical Manager, Aeroelastic Group, Structural Mechanics Division, Air Force Flight Dynamics Laboratory, Wright-Patterson AFB, Ohio. His duties include planning, establishing and directing research and development programs in the field of prediction and prevention of aeroelastic instabilities in flight vehicles. He received a Bachelor's Degree in Aeronautical Engineering from Wichita State University, and a Master's Degree in Aeronautical and Astronautical Engineering from Ohio State University. Mr. Cooley has worked at The Boeing Company, Wichita, Kansas, and has been with the Air Force research and development laboratories at Wright-Patterson AFB, Ohio since 1955. Mr. Cooley is a member of the American Institute of Aeronautics and Astronautics and the Scientific Research Society of North America.

MAXIMUM PERFORMANCE EJECTION SYSTEM

BY

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MAXIMUM PERFORMANCE EJECTION SYSTEM

Abstract

The Maximum Performance Ejection System (MPES) is a development program whose major objectives are: (1) to provide a lightweight, high performance ejection seat through the application of new concepts and techniques; (2) to provide a system with minimum maintenance requirements and maximum reliability and (3) provide a system with the capability to recover from extreme adverse attitudes, a capability which is presently nonexistent. The system itself consists of: a lightweight seat structure made of bonded aluminum honeycomb; a microprocessor controlled timing and sequencing subsystem to provide accurate timing of the propulsion, control, stabilization and recovery events; a long duration spherical rocket propulsion system; a positive positioning and restraint system; a new pilot chute and extraction rocket motor; a new recovery parachute and packaging technique; a seat stabilization and steering system capable of performing a vertical seeking maneuver; a chemically generated emergency oxygen system; and an adjustable crewman headrest. Feasibility components have been fabricated and demonstration tests have been performed.

INTRODUCTION

The Maximum Performance Ejection System (MPES) program was initiated in 1969 by the Naval Air Systems Command for the purpose of addressing the various problems associated with escape systems in modern aircraft. The advent of high performance and V/STOL type aircraft has brought about the need for dramatic improvements in aircrew escape systems since the operational requirements of these aircraft often result in flight conditions which exceed the performance envelope of current escape systems. Typically, escape from an inverted aircraft with present systems requires minimum terrain clearance of 200 to 600 feet. Current ejection seats are also limited in countering the effects of high sink rates when escape is initiated close to the ground. Seat propulsion, timing and the sequencing of systems cannot be optimized over the entire ejection band since the best performance under one set of conditions may give degraded performance for a different set of ejection conditions. In addition, maintenance procedures required in the numerous types of ejection seats mandates specialized training of personnel and excessive manning levels. This maintenance problem is further aggravated by the present day recovery systems which require ground support paraloft facilities and frequent and costly parachute repack cycles. With the Department of Defense particularly interested in decreasing equipment maintenance expenditures, it becomes increasingly necessary to have a highly reliable, low maintenance, light weight escape system. The MPES program, therefore, is directed toward developing an escape system with superior performance over an expanded operational envelope, with minimum maintenance requirements, while paying particular attention to the attainment of high reliability. The overall thrust of the program is to significantly advance the state-of-the-art of escape system technology in each of these areas.

The objectives of MPES are being met under the management and direction of the Naval Air Systems Command and a Navy Engineering Team composed of the following participating Naval activities:

NAVAL AIR DEVELOPMENT CENTER
NAVAL WEAPONS CENTER
NAVAL PARACHUTE TEST RANGE
NAVAL ORDNANCE STATION
NAVAL AIR TEST CENTER
NAVAL WEAPONS ENGINEERING SUPPORT ACTIVITY

MPES DESCRIPTION

The Maximum Performance Ejection System consists of the following major subsystems:

Structural Subsystem

The Structural Subsystem illustrated in Figures No. 1 and 2 consists primarily of a seat back with rails and a bucket for housing the seat Propulsion and Control Subsystem and a seat lid mounted on top of the bucket. Unlike other seats which house the recovery subsystem within a head box or in back of the crewmember, the area between the guide rails of the MPES has been reserved for the Recovery Subsystem in addition to other key components. Contoured space on the front surface of the seat back has been allocated for a "soft pack" survival kit which is a departure from the present day heavy and cumbersome Rigid Seat Survival Kit (RSSK). The seat structure is fabricated of aluminum honeycomb sandwiched between two aluminum sheet facings. The core thickness varies from 3/8 inch to 3/4 inch and the facings range from 0.012 inches to 0.020 inches to provide the most efficient structure. Weight of the basic structure is approximately 25 pounds with a demonstrated structural strength capability of meeting MIL-S-18471 crash safety criteria. The aluminum honeycomb structure allows the internal routing of control cables and electrical lines for maximum shielding and protection.

The following tests have verified the capabilities of this design concept:

a. 44 G combined crash load tests (Philadelphia Naval Base, Feb 1970)
b. 20 G ejection test (Ejection Tower, Philadelphia Naval Base, Mar 1970)
c. 119 ballasted ejections at 10 to 12 G's and at G onset rates of 250 to 1800 G/sec. (Ejection Tower, Philadelphia Naval Base, Jan 1972 to Oct 1972)
d. 51 live subject ejections for high G study (Ejection Tower, Philadelphia Naval Base, Jan 1972 to Oct 1972)
e. 150 knots dynamic ejection (Lakehurst, Sep 1971)
f. 2 static ejections - TVC-SI (Thrust Vector Controlled-Secondary Injection) (Lakehurst, Feb 1973 and Mar 1973)
g. 1 static ejection - TVC-GM (Gimballed Motor) (China Lake, May 1973)

Survival Subsystem

As seen in Figure No. 1 the survival pack is mounted on the seat back in the lumbar region of the occupant. The "soft" pack contains, in addition to all the conventional emergency survival equipment, an oxygen accumulator for emergency breathing. Emergency oxygen is supplied to the accumulator by a Chemical Oxygen Generating System, thereby eliminating the need for interfacing with Liquid Oxygen stores aboard air-capable ships and risking the hazards associated with frequent checking and maintenance.

Restraint Subsystem

The Restraint Subsystem is composed of a series of passive and active components working together to keep the crewmember securely restrained in the seat during emergency egress but unimpeded during his normal mission performance routines. It consists of a contoured and adjustable headrest, extending seat side panels, ballistic inertia reel, inertia reel strap cutter, automatic lap belt cinching and release devices, an inflatable head restraining collar, and a leg restraint and positioning system.

The headrest acts to support the head during normal takeoff and flight conditions. For catapult takeoff, the headrest is manually adjusted to its full forward position to support the crewman's head. In this position, the crewman can keep his eyes focused on the instrument panel displays during the catapult stroke. During flight the crewman can adjust the headrest for his own personal comfort. Upon initiation of ejection, the headrest automatically retracts into the full aft position to ensure proper head alignment for the ensuing upward acceleration.

An inflatable neck bag is used in conjunction with the headrest to prevent head rotation or whiplash due to ejection forces. The neck bag is stored on the crewman's flight suit collar and in its deflated state is little more than a fold in the garment. Upon ejection, gas is introduced into the bag, unfurling and filling it so that it encompasses the crewman's neck, bearing against his chin and supporting his head in an upright position. It remains inflated through the ejection sequence and subsequent parachute opening jolt. During parachute descent the crewman has the option of removing it to allow freedom of head movement.

The inertia reel is a qualified device. The two straps

exiting from the reel are connected to a standard torso harness. At the time of ejection the inertia reel is gas actuated and retracts the crewman's shoulders back to the seat as part of the spinal alignment procedure. The lap belt cinch and release device are actuated by the same gas source as the inertia reel. The cinch operation retracts the lap belt to anchor the crewman's hips and to prevent "submarining" upon ejection. Both the inertia reel and lap belt straps are released automatically as part of the seat/man separation sequence.

As an option, the powered inertia reel can function in a recyclable mode to provide the crewman with a powered and controlled retraction within the cockpit. This system permits the pilot to regain proper seating position in the event he is thrust out of his seat by adverse flight conditions or out of control spins.

Recovery Subsystem

The MPES Recovery Subsystem takes advantage of the large amount of free space available between the seat guide rails. Its placement on the smooth back surface of the seat gives the best assurance that it will not entangle with seat structure. It consists of a seat back mounted Pilot Chute Extraction Motor (PCEM), a sealed pilot chute and a vaccuum packaged personnel parachute system. The Recovery Subsystem will provide the pilot with the optimum in reliability through the use of a microprocessor control system to precisely time the sequence of events of the sealed pilot chute and a reefed 28 feet flat circular canopy.

The STORM (Sealed TO Reduce Maintenance) parachute packing concept developed by the National Parachute Test Range permits the parachute to have a 5 to 7 year service free life and thus eliminates the requirement for a paraloft aboard air capable ships, frequent (217 day) repack cycles and costly logistics and manning support.

Propulsion and Control Subsystem

This subsystem, located under the seat and utilizing a major portion of the seat bucket, provides the MPES with a unique seat steering and stabilization capability. This capability dramatically expands the safe ejection envelope of the escape system.

Statistics from the Naval Safety Center, Norfolk, VA. noted in Figure No. 3, clearly show the trend in survival

rates associated with aircraft ejections. In identifying the major causes contributing to this trend, Figure No. 4 illustrates the most significant of these as being caused by ejections initiated out of the safe ejection envelope.

In an effort to increase the survival rate in this area, a program was initiated to develop a seat propulsion and control subsystem which would provide MPES with an expanded operational envelope and enable it to perform a vertical seeking maneuver. In order to develop this capability, it was planned to utilize, to a maximum extent, the thrust vector control and guidance technology developed for missile applications. A series of three seat flight tests was planned to demonstrate the concept.

At the start of the program, three alternatives for a vertical-up reference were considered. A system using the voltage potential and polarity of the electrostatic field surrounding the earth offered the advantage of being fully independent of aircraft avionics and thus not affected by a failure in that area. Preliminary investigation and tests indicated that although the concept appeared to have potential, the research and development costs were beyond the scope of the program. A second alternative was a system that was initialized by the aircraft attitude reference system at the time of ejection. Successful operation of the escape system in the vertical seeking mode was, quite obviously, dependent on the aircraft attitude reference being functional when the emergency occurs. A third alternative was incorporation of a continuously operating, self erecting attitude reference onboard the seat. Cost and maintenance requirements appeared to make the third alternative an unattractive approach. The second alternative was selected as the most viable approach at that time. A digital autopilot designed around an Intel 8080A microprocessor and strapdown electro-mechanical rate gyros were selected to implement the system. The microprocessor also served as an accurate and flexible sequencer for functions such as parachute deployment and restraint release.

It was recognized at the beginning of the program that propulsion requirements for a vertical seeking system would be significantly different than those for contemporary systems. A system simulation was assembled to evaluate propulsion performance and control system stability. Several variations of thrust/time curves were tried. A selection was made based on minimum loss of altitude during an inverted attitude ejection. Physiological limitations on

accelerations and rotation rates were also considered. Available aerodynamic data was not adequate to evaluate yaw control requirements, where yaw is defined as rotation about the Z-axis using the aircraft coordinate system convention. However, yaw attitude did not present a problem during the zero airspeed conditions of the planned tests as the air velocity vector was quite closely aligned with the Z-axis. Yaw rates induced by products of inertia were not considered high enough to interfere with parachute deployment and seat/man separation. An 8" diameter spherical rocket motor configuration was selected. The rocket was mounted in a two axis gimbal arrangement beneath the seat pan allowing ± 16 degrees TVC (Thrust Vector Control) in the pitch and roll planes.

Gimbal position was controlled by hydraulic actuators and off-the-shelf Hydraulic Research Model 25A servovalves. Hydraulic power was supplied by a blow down accumulator. An explosive valve, controlled by the sequencer, opens to provide pressure to the system when ejection is initiated.

Preliminary testing consisted of a static rocket motor firing, bench tests of the actuation system and autopilot, and actuated rocket motor firing and flight simulator tests. During flight simulator tests, the rate gyros were mounted on a three axis flight simulator (Carco Table) representing the seat. Seat dynamics were programmed in an analog computer connected to the Carco Table. An inert rocket motor and actuation system were controlled by the autopilot. During simulation of the planned flight tests, rocket motor position and rate and seat position and rate were monitored. System gains were adjusted to their final values during flight simulation tests.

The first of the series of seat flight tests was a zero-zero test. The system did not control and parachute deployment did not occurr. It was later determined that a short in a check-out switch had caused a printed circuit line to fuse resulting in a loss of electrical power. The problem was corrected and a repeat test conducted. Seat attitude stabilized well during this test. Two anomalies occurred: the system was deliberately flown with approximately 2" of thrust/C.G. misalignment. It was anticipated that an attitude error signal of 26 degrees would be required to achieve the necessary TVC angle. A seat attitude error of 54 degrees occurred. Examination of telemetry data indicated signal clipping occurred, effectively reducing system gain. It was later determined that a natural seat frequency had excited the rate gyros,

resulting in the signal clipping. The second anomaly occurred when the pilot chute failed to open the vaccuum packed personnel parachute. A safety interlock prevented seat/man separation and the system was recovered under the seat recovery chute.

The seat was modified to raise the natural frequency, and filters were installed in the autopilot. The personnel parachute container was modified to facilitate opening from any angle of pull. The A-6 cockpit was suspended 100 feet Above Ground Level (AGL) in a 90 degree roll attitude for the following test. The test was successful in all respects. The seat dropped approximately 2.5 feet below the ejection level and then climbed to 486 feet AGL. Personnel parachute deployment, seat/man separation and seat recovery parachute deployment all occurred in a normal manner.

The final test was conducted with the A-6 cockpit suspended in a fully inverted attitude at 100 feet AGL. The vertical seeking maneuver was initiated at rail separation. Lowest point on the trajectory was 57 feet AGL, at which point the seat achieved an upright attitude. Rocket motor burnout occurred just below the ejection point. The seat continued to 150 feet AGL as the pilot chute and personnel chute deployed. Seat and dummy were successfully recovered under the personnel parachute.

Several seat attitude reference concepts have evolved during the feasibility demonstration program. Among these are a system in which rate sensors operate continuously, but are updated at frequent intervals by the aircraft. In the event of a failure of the aircraft system, the seat system would retain an adequate reference for a substantial length of time. Recent developments in rate sensors, such as the vibrating wire sensor, appear to offer high reliability and long life, characteristics which make the above concept feasible. Another concept involves using Microwave Radiometry (Micrad) to sense seat attitude. Preliminary investigations appear promising. The system is passive, independent of the aircraft, and economical to implement. It appears that rate as well as attitude can be obtained from the signal. A program is planned to demonstrate feasibility during FY 79.

The vertical seeking concept will not only greatly increase the escape envelope, but will also allow a "softer" ride and a slower, more controlled parachute deployment thus decreasing the incidence of injuries.

SYSTEM DEVELOPMENT PLAN

The development plan for the MPES program can be seen in Figure No. 5 with a target completion date of 1984. The Maximum Performance Ejection System will be available for introduction to the fleet at that time.

To assist in the advanced development and engineering development phases of this program the Navy will solicit industry inputs for system integration and development. It is anticipated that contractual committments for this work will be made in Fiscal Year 1981.

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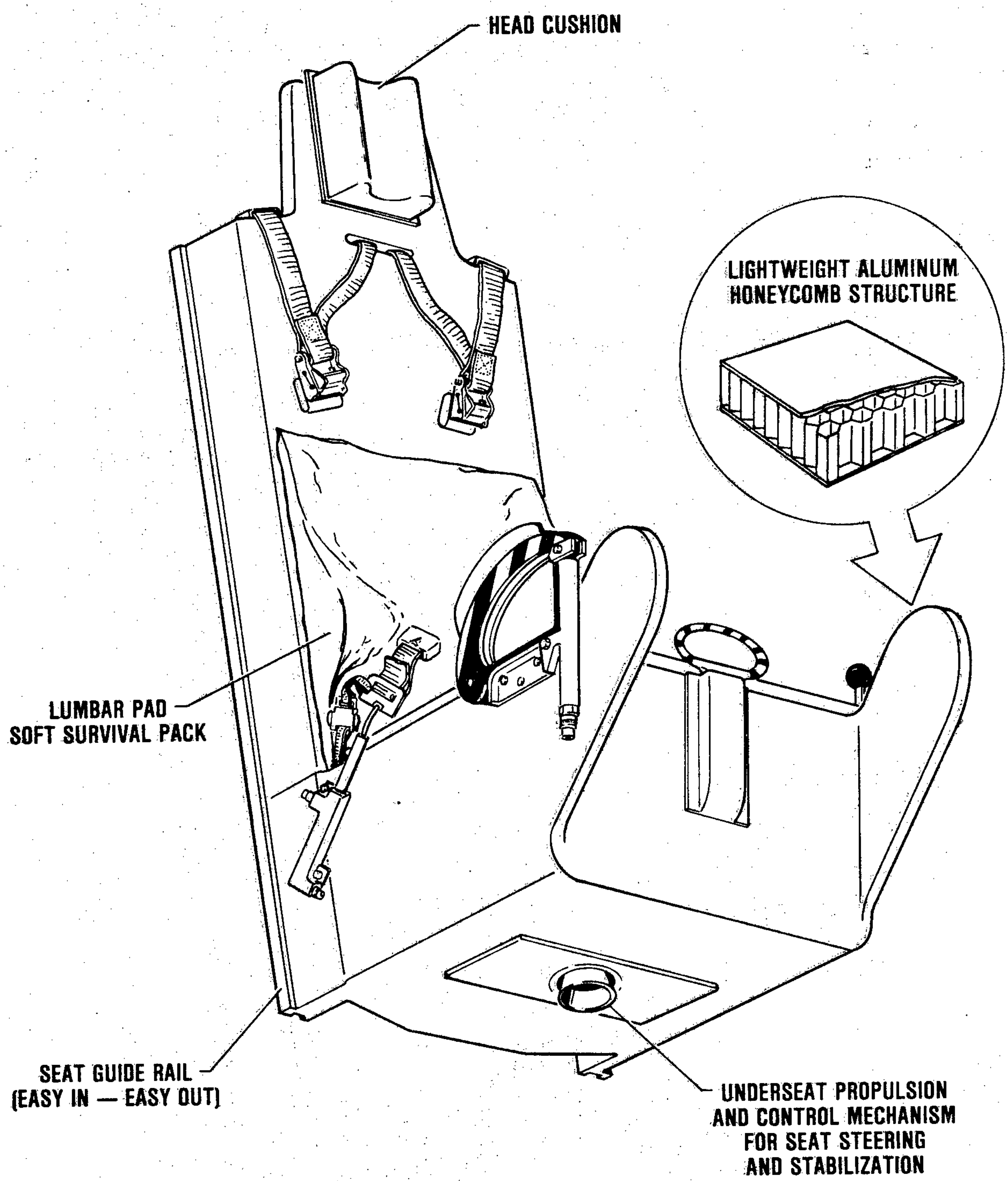

FIGURE 1. MAXIMUM PERFORMANCE EJECTION SYSTEM (MPES)

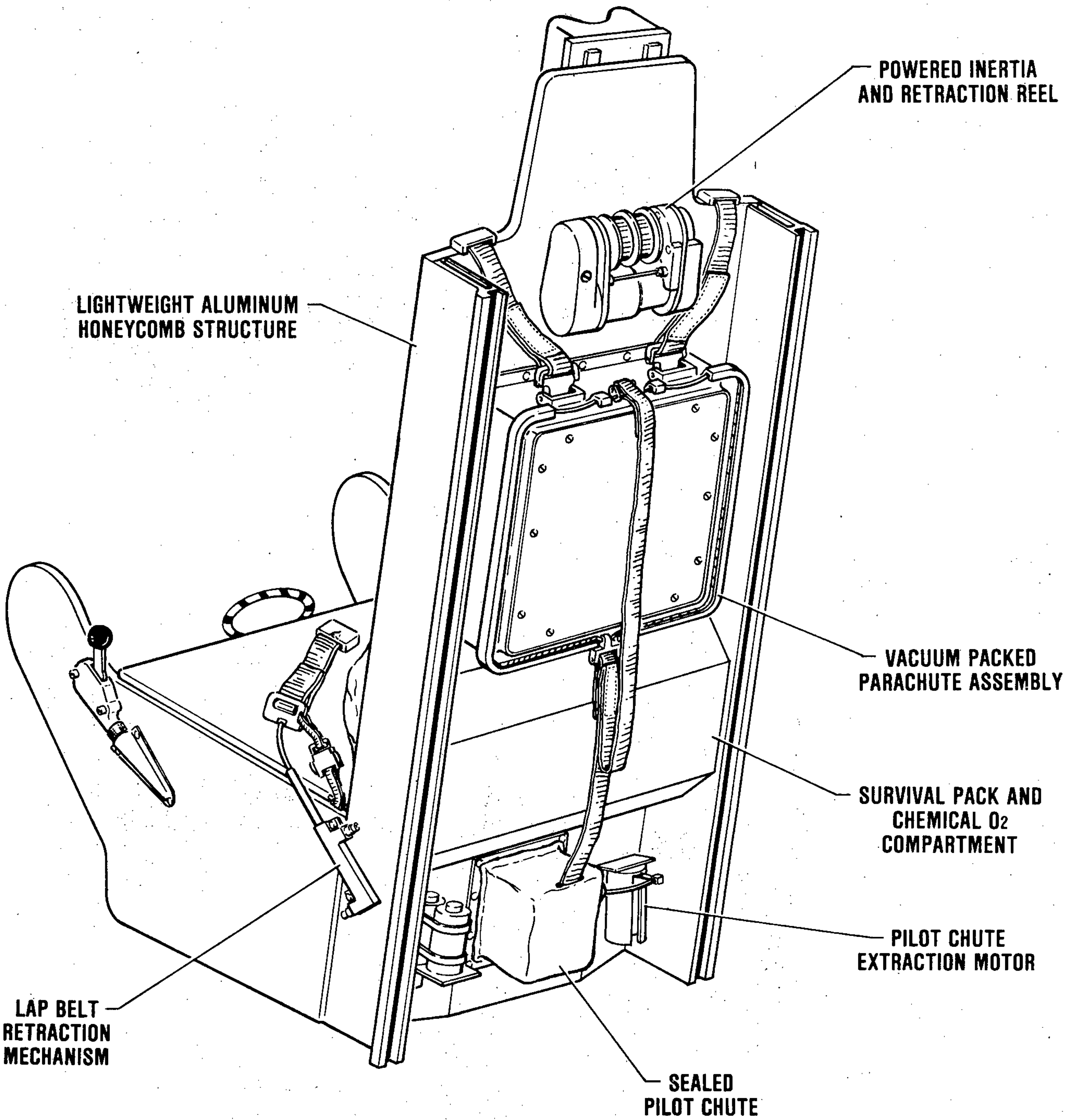

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FIGURE 2. MAXIMUM PERFORMANCE EJECTION SYSTEM (MPES)

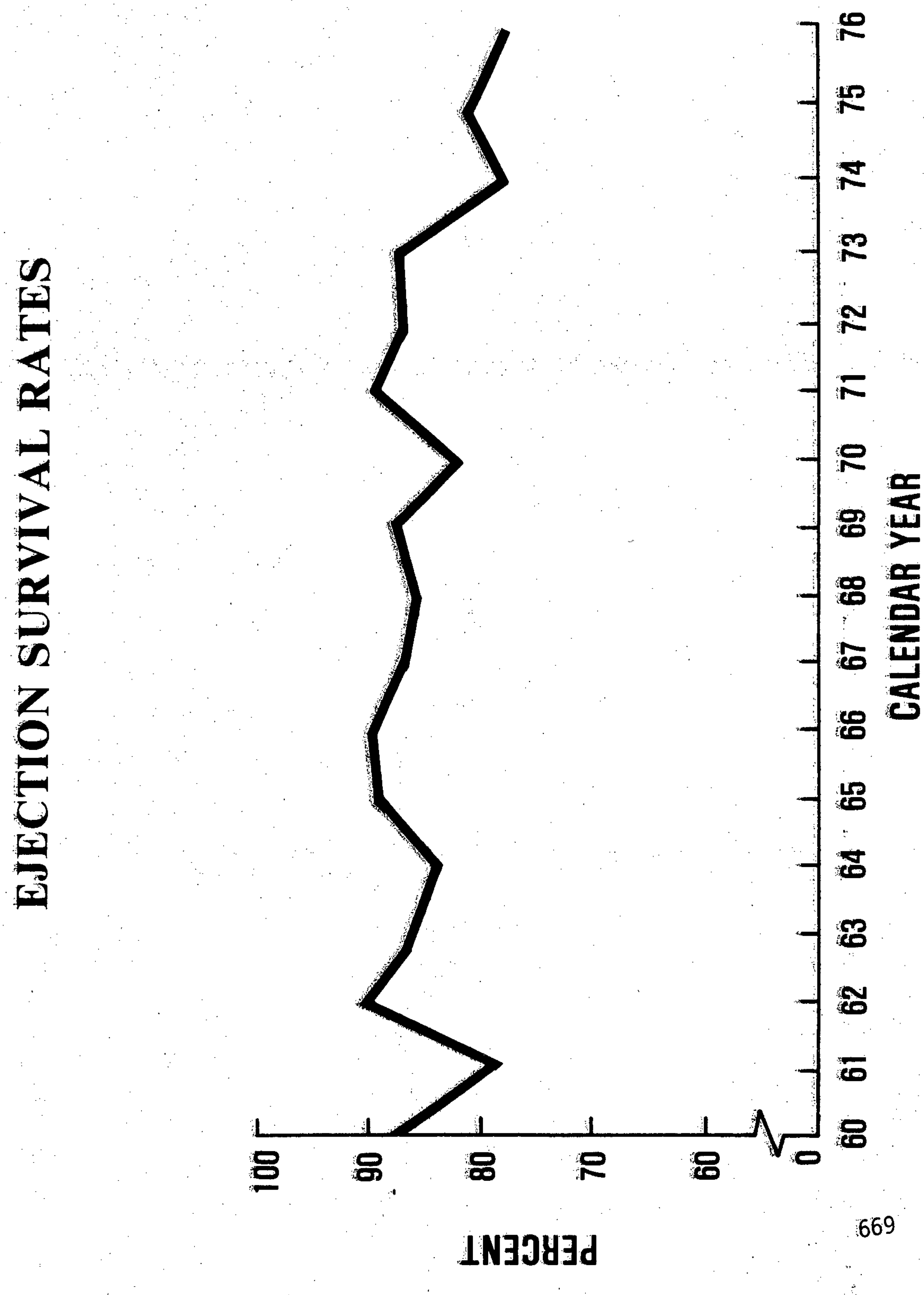

FIGURE 3. EJECTION SURVIVAL RATES

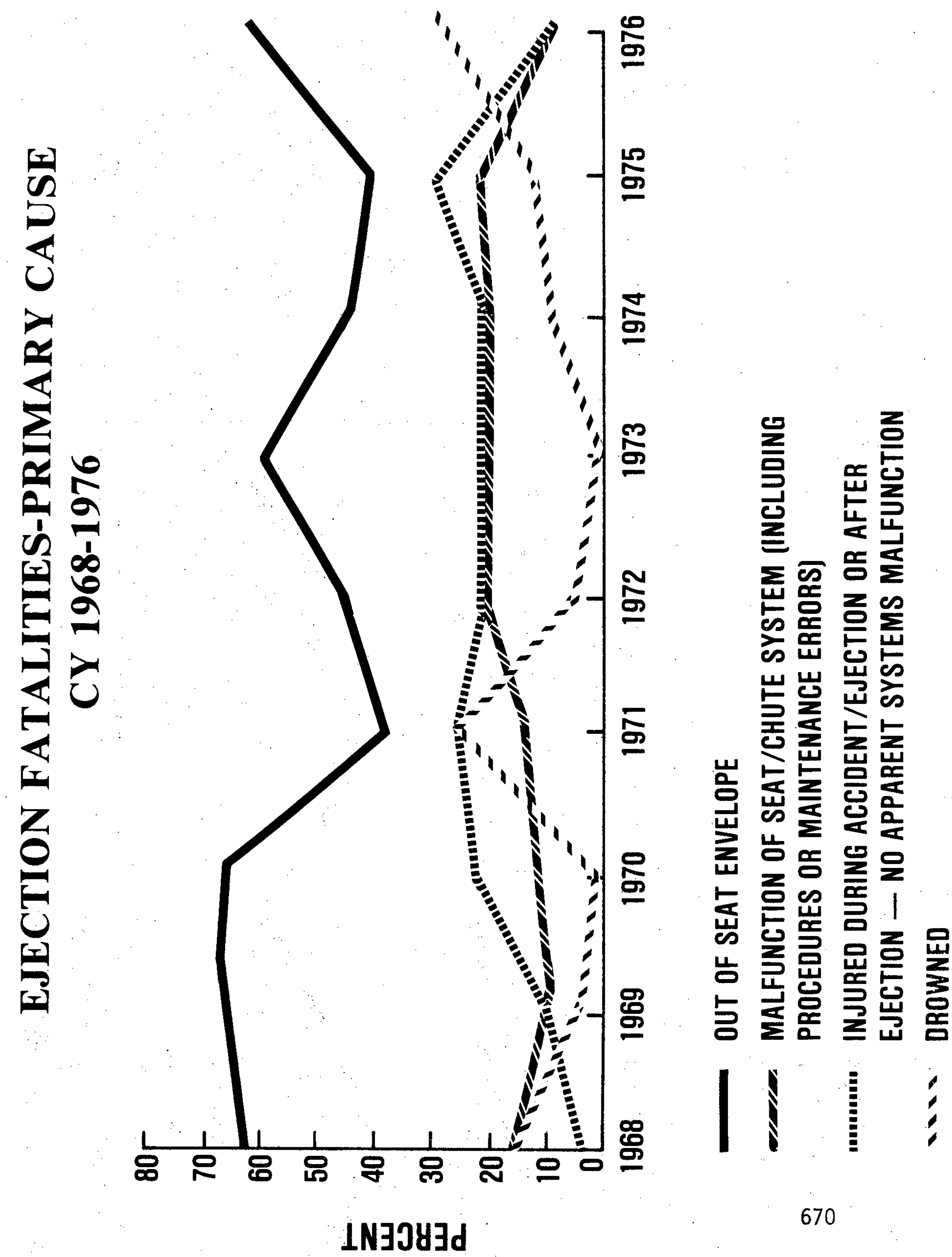

FIGURE 4. EJECTION FATALITIES-PRIMARY CAUSE CY 1968-1976

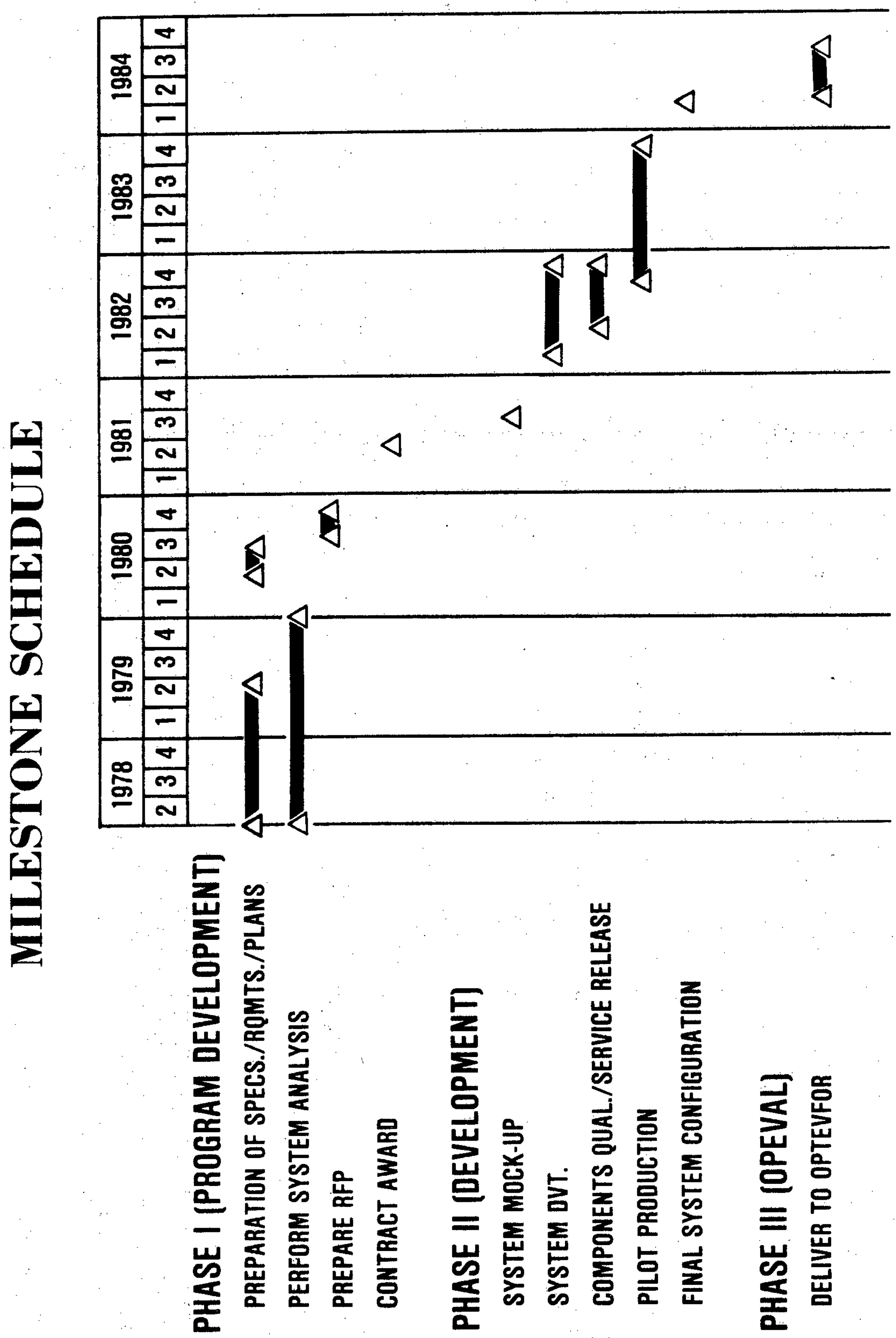

FIGURE 5. MILESTONE SCHEDULE

Biographical Sketch

John J. Tyburski

Mr. Tyburski is presently an Aerospace Engineer for the Crew Systems Division of the Naval Air Development Center. He received an A.A.S. degree in Mechanical Technology from the Academy of Aeronautics, New York and a B.S. in Aeronautical Engineering from Tri-State University, Indiana. Mr. Tyburski's experience has been primarily in escape system research and development and he is currently the Project Engineer of the Maximum Performance Ejection System Program.

W. J. (Bill) Stone

Mr. Stone graduated from California Polytechnic State University, San Luis Obispo, CA with a B.S.M.E. in 1954 and is registered as a Mechanical Engineer in the State of California. Mr. Stone has been employed at the Naval Weapons Center since 1962. A major portion of that time has been spent on design and development of escape system propulsion.

STATUS OF CIRCULATION CONTROL ROTOR AND X-WING VTOL ADVANCED DEVELOPMENT PROGRAM

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ABSTRACT

The current status of circulation control aerodynamics research and of two full scale flight demonstrator aircraft development programs, employing circulation control rotors, is presented. The development of circulation control technology has taken an orderly progression from two-dimensional airfoil experiments and analytical studies through three-dimensional subscale rotor experiments and correlation to full scale rotor development and flight tests. The first program, the Circulation Control Rotor (CCR), includes the installation of a conventional speed circulation control rotor on an existing H-2D airframe. This development is being pursued under U.S. Navy contract to Kaman Aerospace Corporation, Bloomfield, Connecticut. The second program consists of the development of a new VTOL aircraft, the stopped rotor X-Wing. This flight demonstrator is being developed under DARPA (Defense Advanced Research Projects Agency) contract to Lockheed Aircraft Corporation, Burbank, California. Both of these programs are scheduled to be tested in the NASA Ames 12 x 24 m (40 x 80 ft) wind tunnel during late 1978.

In-house laboratory research programs leading to the translation of CCR and X-Wing technologies into industry are described. Current status of the CCR design, fabrication and initial whirl stand validation efforts, in preparation for the July 1978 tests in the Ames tunnel, are described. Plans call for testing the flight vehicle early in 1979. Also described are the results of a preliminary design study and hardware fabrication program for the X-Wing. The X-Wing rotor is to be extensively tested on the whirl stand in preparation for NASA wind tunnel tests around September 1978. The results of current theoretical and experimental analyses of all flight modes of both the CCR and the X-Wing show that no fundamentally limiting problems exist at the present stage of development.

1. INTRODUCTION

The Circulation Control Rotor (CCR) and the X-Wing are each the integrated product of several advanced technologies which have reached maturity only within recent years. The use of circulation control (CC) aerodynamics is the primary innovation which permits a radical departure from the limitation of classical sharp trailing edge airfoils. The quasi-elliptical CC airfoils utilize a unique form of boundary layer control over a rounded trailing edge to produce lift substantially independent of both incidence and velocity. The major aerodynamic limitations of higher speed helicopter flight – retreating blade stall, reverse flow and advancing tip Mach number – may be effectively alleviated by the CC airfoil. Analyses have shown that CC rotor systems should have inherently lower vibration levels due to the 'velocity independence' characteristic of circulation control lift. An important aspect of a CC rotor is the

pneumatic control system which is inherently simple while also enabling incorporation of higher harmonics of control input to further reduce blade stresses, transmitted shears, vibration and power requirements. The entire hub and control system can be conveniently faired to virtually eliminate the hub/pylon drag.

A second major technology incorporated in both these programs is the integration of carbon fibre composite materials within totally new rotor blade structures. This combination allows the blade designer to exploit the advantages of rigid rotors with levels of stiffness-to-weight ratio which heretofore were considered impossible. In the case of the CCR, the resultant blade design precludes rotor mechanical stability problems and assures stable flutter-free rotor operation over the design flight spectrum. For the X-Wing, such historical stopped-rotor limitations as aeroelastic divergence, flutter, blade dynamic instability, and critical resonance conditions during rpm reduction are alleviated.

One final advanced technology in the case of the X-Wing is a unique method for stabilizing the rigid rotor by sensing the rotor shaft moments, segregating out those which are extraneous to the pilot input, and feeding back corrective pneumatic signals to cancel out gusts and other disturbances. The specific implementation of this 'moment feedback control system' in the X-Wing permits the same hardware and software to be used in all flight modes with no more than simple gain changes.

2. DESCRIPTION OF CONCEPTS

In principle, both concepts utilize a shaft-driven rotor with blades having circulation control airfoils (Figure 1). The CC airfoils employ a rounded trailing edge with a thin jet of air tangentially ejected from a slot adjacent to the (Coanda) surface. The jet of air suppresses the boundary layer separation and moves the rear stagnation streamline toward the lower surface, thereby increasing lift in proportion to the duct pressure. Pitch and roll control requirements are obtained by cyclic modulation of the compressed air which is ducted through each blade. A simple throttling mechanism is used in the rotor head to provide both the cyclic and collective components of blown air, thus providing the required cyclic and collective rotor control. This process eliminates the need for blade cyclic pitch change and may also eventually eliminate the need for blade collective pitch. (The latter would be dependent on the aircraft hover requirement and 'engine out' capability.) The rotor head is, therefore, free of numerous dynamic control system components (directly related to reduced maintenance and increased reliability), thus greatly simplifying the mechanisms while presenting a cleaner (reduced drag) profile. This is the basic concept of the Kaman CCR.

As helicopter speed increases conventional rotor blades stall due to the combined effects of reduced dynamic pressure on the retreating side of the disk and the relatively low maximum lift coefficient of classical airfoils. Because of this fundamental limitation, conventional rotors exhibit strongly reduced lift capability (while maintaining trim conditions) as the advance ratio increases. Several "fixes" to this problem have been attempted (such as increased solidity ratio, auxiliary wings, and contra-rotating rotors) but each of these results in increased weight, complexity and dynamic problems. With circulation control higher speeds and advance ratios are achieved using a second duct and leading edge slot (Figure 2) so that the rotor can develop significant lift in the region of reverse flow. This is the basic 'reverse blowing' method which is fundamental to the X-Wing concept. Two-dimensional airfoil experiments have shown it is possible to develop large lift coefficients by blowing from the

appropriate individual slot or from both slots simultaneously. The advancing blade employs only trailing edge blowing because the relative velocity in this region is always in the conventional direction. The dual blowing concept on the retreating side allows the X-Wing rotor to maintain relatively high rotor lift and moment trim capability in the conversion flight range ($0.5 \leq \mu \leq 1.4$) where the retreating blade experiences mixed flow conditions (i.e., locally reversed flow on the inboard sections and forward flow on the outboard sections).

3. DEVELOPMENT OF CIRCULATION CONTROL TECHNOLOGY

The Circulation Control Rotor is a byproduct of research efforts conducted in the early sixties under the direction of Dr. I.C. Cheeseman at the National Gas Turbine Establishment in Great Britain. This work established the feasibility of the basic circulation control concept for rotor systems, Reference 1. Extensive development of the concept including two-dimensional airfoil analyses and testing, subscale CCR wind tunnel model rotor testing, high advance ratio reverse blowing rotor testing, and development of advanced rotor theory has been continuing since the late sixties at the David W. Taylor Naval Ship Research and Development Center (DTNSRDC). This work established the technology base and provided the impetus for the United States government to involve private industry in the development of the two full scale rotor programs described here.

Early work at DTNSRDC supported under Internal Research and Exploratory Development (IR/IED) funding concentrated on the fundamental aerodynamic mechanisms of the CC airfoils.* Significant improvements in blowing efficiency were obtained after intensive in-house analytical and experimental efforts which led to the development of a series of advanced CC airfoils, References 2 through 5. Various applications of these airfoils have been studied including rotors, wings, whip propellers, low pressure ratio lift fans, and hydrodynamic control surfaces. (Reference may be made to 'Circulation Control – An Updated Bibliography of DTNSRDC Research and Selected Outside References,' Reference 6 for a partial listing of the more than 50 technical publications.)

Initial CC rotor research at DTNSRDC was concentrated on a four-bladed model rotor designed based upon several years of two-dimensional airfoil and pneumatic valve research. This first model, designated the Higher Harmonic Circulation Control (HHCC) rotor was evaluated at DTNSRDC in 1971, Reference 7. Design requirements of the valving system for the HHCC model, in addition to providing the basic trim function, called for higher harmonic pneumatic control and hub control system simplicity. A cam-type pneumatic valving system was developed to provide helicopter rotor control and moment trim. It was demonstrated, with the use of a breadboard model, that the valving system was capable of providing both first and second harmonic rotor control, Reference 8, by superposition of a constant, a one-per-rev and a two-per-rev pressure signal, resulting in an optimum combination of pressure wave components. This exploratory technology was the basis for the valving systems later used in the low speed Circulation Control Rotor models and eventually in the full scale flight demonstrator (XH2/CCR). The HHCC model proved that trimmed flight could be achieved

*This section on the chronological development of the circulation control technology is pictorially presented in Figure 3. The discussion will follow, moving from the left to the right in Figure 3, the earliest to the most recent technological developments of circulation control.

without moving parts other than the rotating blades. This first DTNSRDC model rotor also demonstrated that the very high lift capability and efficiency of circulation control airfoils could be achieved in the three-dimensional regime. Correlation of the HHCC performance in both hover and forward flight resulted in an improved predictive theory. Variables such as slot height-to-chord ratio, slot height-to-trailing edge radius ratio, Reynolds number, and Mach number were identified as factors which significantly influence the performance of circulation control airfoils.

As the research progressed the availability of refined two-dimensional airfoils and improved rotor theory led to a second generation rotor design, denoted the Circulation Control Rotor (CCR), Reference 9. This two-bladed rotor showed substantial improvement in performance including a twofold increase in rotor lift augmentation. During this period the design was used in preliminary design contractural efforts to assess full-scale feasibility for the CCR concept. Two possible full-scale structural designs and manufacturing processes for the unique geometry of the slotted blade were established in these studies, References 10 and 11. Full-scale pneumatic control systems were also designed and integrated with the CCR stability and control characteristics in order to provide the "control simplicity" philosophy associated with CCR. From these studies it was concluded that the construction of a full scale CCR helicopter was feasible and practical, that the use of CC blowing allows a significant reduction in rotor complexity and that significant (50 percent) reductions in vibratory airloads should be achievable. As a direct result of the model rotor tests and contractor verification of full-scale feasibility, a four-year program was initiated to design, manufacture, and flight demonstrate a full-scale CCR including all associated compressor and control system hardware. The least cost approach to proving the CCR technology in flight was the selection of a retrofited Kaman H-2D airframe, designated the XH2/CCR.

In 1972, design work was begun on a high speed version of the circulation control rotor concept, Reference 12. In-house research was first concentrated on the valving and airfoil technology areas. A new valving system was required to provide, in addition to the collective and cyclic blowing control of the CCR, the capability for individual blowing to the dual slotted blade ducts and also variable azimuthal programming of the airflow. An initial high speed rotor control valve system was evaluated using a breadboard model to verify the control concept. The results of the breadboard model tests, Reference 13, indicated some valving problems associated with the use of the dual slotted blades. Using a combination of empirical and analytical results, the control system was successfully redesigned to remedy these deficiencies enabling the desired pressure wave shape to be programmed throughout all rotor flight regimes. The valving system manufactured for the high speed rotor model, designated the Reverse Blowing Circulation Control Rotor (RBCCR), was based on the breadboard configuration (Reference 14). Concurrent with the development of the RBCCR valving system a new CC airfoil employing blowing slots in both the leading and trailing edges was designed and tested. Two-dimensional experiments demonstrated that this airfoil could develop large lift coefficients even when simultaneously blowing from both slots. Although some lift reduction was evident with dual blowing, high C_ℓ capability was still present (Reference 15).

Based on the results of the valving tests and the two-dimensional dual-slotted airfoil tests, the 6.7 ft (2.04 m) diameter RBCCR was designed, manufactured, and tested in both hover and forward flight, Reference 14. The first evaluation of this model was conducted in 1975 and demonstrated that the reverse blowing circulation control concept was viable and merited

further development. The RBCCR model demonstrated a capability to develop efficient trimmed flight from hover through transition and into cruise to advance ratios of at least 4.0. In hover a power increase due to the open leading edge slot gave a corresponding reduction in hover Figure of Merit. (The reduction in Figure of Merit dictated that a flexible slot approach should be incorporated into later designs to close the unblown slot; fortuitously increased performance in forward flight is an additional advantage of using flexible slots.) In summary, the RBCCR model showed good performance in hover, very good thrust and trim capability at high advance ratios ($0.5 \leqslant \mu \leqslant 1.4$), and excellent lifting system efficiency.

The results of the initial RBCCR model tests, together with the incorporation of the other advanced technologies alluded to earlier, provided the needed ingredients for the evolution of the X-Wing stoppable rotor concept (Reference 16).

The RBCCR model was extensively evaluated during 1976-1977 to investigate stopped X-Wing performance, the effect of shaft angle on rotor 'conversion' performance, and also specialized research into three-dimensional circulation control aerodynamic phenomena such as yaw effects, maximum lift, and tip vortex formation. Test data were correlated against high speed performance prediction theory, Reference 17. It was determined for the conversion flight regime that simple momentum or distributed momentum inflow models were not adequate but that a prescribed wake inflow method was required. This combined theoretical and experimental approach has proven satisfactory correlation with experiment for design of the full scale wind tunnel configuration of the X-Wing.

The present full scale X-Wing control valve design is different in several important respects from the control system used in the previous circulation control models and the RBCCR model. The full-scale control valve consists of an eight port segmented system feeding into blades with flexible slots whereas the RBCCR model control system employs a continuous (infinite port) arrangement feeding into blades with fixed slots. The eight port system, while more practical for a flight vehicle, has limitations in control input for both trim and higher harmonic inputs. The analytic complexity necessary to correctly model the valving-blade internal pneumodynamics and other analytical limitations (primarily due to the uncertainty of the rotor inflow models) in the rotor performance theory raises significant questions about the accuracy of the analytical calculation of rotor trim settings and rotor stability and control derivatives. To supplement the analysis, a new RBCCR model control system was designed and manufactured. This model was evaluated in the spring of 1978, and has provided excellent data on the X-Wing control limits, control derivatives, effects of second harmonic input on rotor performance and blade loads, and basic rotor aerodynamic transfer functions. The data obtained from this series of tests is directly applicable to the full scale X-Wing wind tunnel model.

As a result of the very promising RBCCR model rotor performance and contractor indications of full scale feasibility, a program was initiated to design, manufacture, and evaluate in wind tunnel tests a full scale X-Wing rotor. Upon successful completion of the full scale wind tunnel test a manned flight demonstrator may be designed, manufactured, and flight tested.

4. CCR TECHNOLOGY DEMONSTRATOR VEHICLE

4.1 PROGRAM INITIATION AND OBJECTIVE

With the continual development of CC airfoil technology, the promise of substantial gains in rotor simplicity, and the demonstrated performance from wind tunnel model rotors, it became apparent that the next major step was to demonstrate the CCR concept with full scale hardware. Kaman Aerospace Corporation was selected from an initial list of bidders to build a flight-worthy rotor system of significant size 12,000 lb (5443 kg) such that future vehicle scale-up would be minimal. Kaman's strengths lie in their manufacturing experience, their knowledge of Navy helicopter requirements and the direct approach they took to full scale CCR hardware development. From the earliest hardware development, it was essential to exploit the inherent simplicity of the CCR control system so that substantial reductions in maintenance-man-hours-per-flight hour could be realized. It was also obvious that a major technology breakthrough would be realized to fabricate a rotor blade of high torsional stiffness to compensate for the aft suction peak and still maintain an accurately contoured nozzle along the entire length of the blade. To date, Kaman has met both of these challenges and the only major goal left is to flight demonstrate the XH2/CCR in 1979. The following is a description of the major components of the CCR system. A more detailed description is contained in Reference 18.

4.2 CCR FLIGHT DEMONSTRATOR ROTOR SYSTEM

4.2.1 Compressor

The first system hardware that differentiates the CCR concept from conventional mechanical rotor concepts is the need for a safety-of-flight pneumatic source that provides the desired pressure and mass flow to each blade. This, in turn, produces the associated lift variance around the rotor disk. Figure 4 shows the resulting compressor built specifically for the XH2/CCR requirement by Garrett-AiResearch Corporation. This unit weighs approximately 130 lb (59 kg) with a 13 in. (0.33 m) centrifugal compressor wheel. The compressor provides a pressure ratio of 2.2 and a total mass flow of 10 lb/s (4.5 kg/s). Because the XH2/CCR is a technology demonstrator, a 50 percent contingency margin was added to the calculated required compressor power. Variable inlet guide vanes are used to maintain the best possible efficiency because the compressor operates over a wide range of required pressures and associated mass flows. The pilot, through his collective control, has direct input to the inlet guide vanes. From the compressor, the air is pumped through a duct-scroll to the hub plenum where the control valve system is located.

4.2.2 Hub and Controls

In the design of the control valve, the main emphasis was to minimize the number of rotating pieces and yet produce a one-per-rev control capability as well as, at least, a fourth harmonic additional control input. In the rotor hub, four nozzles, one for each blade, pass over a thin nonrotating steel plate that can be moved relative to the nozzle openings to produce the desired throttling. Figure 5 shows the completed flex ring valve. The series of springs and actuators can provide standing wave forms in the flex ring permitting higher harmonic control inputs. The flex ring valve remains stationary, and the control input interface between rotating and nonrotating hardware is simply the throttled air. Less than 1/2 in.

(1.27 cm) deflection of the flex ring is required to produce maximum rotor moments, without any substantial loads as opposed to conventional push rods and swashplates. All hardware associated with the pneumatic control system has been successfully demonstrated during this past year without any major difficulties.

This very simple control can be installed in a hub that requires neither swashplate nor cyclic blade angle-of-attack requirements. The XH2/CCR technology demonstrator has the capability for blade collective angle-of-attack changes to select optimum blade settings for various flight regimes. It might be possible to completely eliminate the collective requirement if sufficient performance during autorotation entries can be obtained. As can be seen from Figure 6, the CCR concept enjoys the simplicity of the hingeless rotor approach, however, sufficient stability and control margins must be maintained.

4.2.3 Rotor Blade

The CCR demonstrator was designed with blade properties as stated in Figure 7 that will minimize the risk associated with developing an advanced rigid rotor.

From early wind-tunnel testing of 6 ft (1.8 m) diameter rotors, it was found that the aerodynamic performance of each is extremely sensitive to slot geometry both in nozzle formation and Coanda trailing edge shaping. Therefore, great attention was given to the laying out of the full scale blade including the position of the slot jacking posts, trailing edge internal contours, trailing edge curvatures, and slot lip thickness. Also, as mentioned previously, the CCR demands high torsional stiffness (10 per rev) requiring the use of composite technology and particularly graphite. The blade has interwoven graphite and unidirectional glass to provide the desired properties. Figure 8 shows the root end section being laid-up with Figure 9 showing the final product. Over twenty blades will be produced during this phase of the program; blade number 12 has been completed at this time. Over 5 million cycles, representing an accelerated fatigue schedule have been completed with a second blade now in preparation for similar testing. To date, no major technical problems have arisen in blade fabrication; however, excellent work habits must be maintained with each blade during lay-up to ensure consistency. All blades are fabricated from the same closed die molds as shown in Figure 10 using standard autoclave processing.

4.2.4 Proof Tests and Wind Tunnel Tests

With the completion this spring of the sixth blade, it was possible to conduct a full scale proof test of the first CCR on the Kaman whirl stand as shown in Figure 11. The results of blade tracking were most gratifying. Maximum out-of-track without blowing was less than 0.25 in. (0.64 cm) and with blowing less than 0.125 in. (0.32 cm) thereby negating the need to adjust the slot height for tracking. Over 17 hours were accumulated on the blades prior to delivery to NASA Ames for full scale wind-tunnel evaluation. Over 10,000 lb (4536 kg) of thrust was generated on the whirl stand by simple angle-of-attack lift and no blowing. Due to limits of the whirl stand hardware and drive system, the maximum thrust that could be generated was slightly over 14,000 lb (6350 kg); however, over 20,000 lb (9072 kg) of thrust are planned in the Ames wind tunnel. Due to the ability to rapidly change the inlet guide vanes, the rotor was unloaded on the whirl stand from 12,000 lb (5443 kg) to 3000 lb (1361 kg) in one sec without any measurable rotor instabilities. Control response was very clean and rapid with rolling moments generated that exceed the current requirements of the

SH-2F in hover. These results are extremely promising with the final verification coming from the data generated at NASA Ames where forward speeds simulating 160 knots can be obtained. Figure 12 is a photograph of the second CC rotor mounted in the Ames wind tunnel.

4.3 CCR SUMMARY

With completion of the wind tunnel program, the CCR will be ready for tie down testing on the actual H-2D airframe. This effort will ensure final compatibility of the rotor and airframe. It is currently anticipated that flight testing can begin this February for a total of 50 hours. The XH2/CCR has been designed for a total of 150 hours thereby resulting in the potential for 100 additional flight hours. With the success of the initial flight program, the effort will move into the next phase of validating the higher harmonic control capability for suppression or elimination of rotor induced vibrations. In addition, a military operational evaluation would be conducted. Current technology development of the basic airfoils and improvements in hardware implementation of the CCR concept offer extremely promising increases in both helicopter performance, reduced vibration levels, and reductions in maintenance levels.

5. X-WING FLIGHT DEMONSTRATOR PROGRAM

5.1 PROGRAM INITIATION AND OBJECTIVES

As previously described, the laboratory program established the aerodynamic feasibility of the dual blowing CCR concept as applied to a stoppable rotor. A related study (Reference 16) expanded the scope of this feasibility investigation to an entire aircraft, concentrating on the other key areas of structural weights, dynamics, and stability and control. The resulting conceptual X-Wing aircraft design exhibited sufficient potential to warrant further detailed exploration by an airframe contractor. Accordingly, during 1975 a one-year contract was awarded to the Lockheed California Company (LAC) by the Defense Advanced Research Projects Agency (DARPA) to perform a preliminary design study of an X-Wing flight demonstrator vehicle. Lockheed was selected because it was known to have a very strong capability in the nonlinear analysis of stiff in-plane hingeless rotors, the automatic stabilization of high advance ratio rotors, and the ability to design and fabricate high technology airframes in minimum time. Much of the accumulated DTNSRDC technology was transferred to LAC at the initiation of the contract. The primary objective of the program was the design of a minimum cost aircraft which could demonstrate the critical technologies of the X-Wing and provide eventual application of the technology to a prototype operational vehicle. The successful completion of the preliminary design study led directly to the current flight demonstrator program (Phase I, wind tunnel; Phase II flight tests) which has the following major objectives:

1. Demonstration of high aerodynamic efficiency and high speeds during fixed wing cruise flight.

2. Demonstration of a low VTOL lift system structural weight fraction when scaled to an operational aircraft size.

3. Demonstration of acceptable flight dynamic characteristics and handling qualities during all flight modes, but particularly during conversion from rotary to fixed wing (and reverse).

4. Correlation of X-Wing analytic models with wind tunnel and flight test sufficient to proceed with confidence to an operational aircraft size.

The first two objectives are the crucial ingredients for the success of any rotary wing VTOL aircraft. The third is critical to the X-Wing insofar as conversion flight is concerned but also encompasses the low speed and hover regimes where severe requirements exist for eventual shipboard operation aboard frigates and destroyers. The last objective is key to the timely and efficient translation of the technology to service use.

5.2 PRELIMINARY DESIGN STUDY

During the preliminary design study effort, the contractor explored essentially all of the key technical questions regarding the X-Wing which had been previously identified by DTNSRDC and a specially convened DARPA panel of technical experts. The resulting configuration for a manned, single seat airplane is shown in Figure 13. Pertinent design data are listed below:

Gross Weight	3200 lb (1452 kg)
Rotor or Wing Diameter	25 ft (7.6 m)
Root Chord	22.67 in. (0.575 m)
Tip Chord	11.33 in. (0.288 m)
Tip Speed	529 ft/s (161 m/s)
Disk Loading	6.53 lb/ft^2 (312.6 N/m^2)

In conjunction with the design effort the study included in-depth analysis of such key areas as the vehicle stability and control during conversion (to and from fixed wing flight), rotor blade and wing dynamic and aeroelastic stability, structural loads, and control valve pneumodynamics. Sufficient blade specimen work was conducted to insure that the blade design concept could actually be built. An important ingredient in the successful conclusion of this study was the establishment of a very small closely knit team of top technical professionals from both the contractor and government organizations.

5.3 FABRICATION, PROOF TEST, AND WIND TUNNEL TESTS OF THE FLIGHT DEMONSTRATOR ROTOR SYSTEM

Although an extensive quarter-scale model test program has been conducted at DTNSRDC with both rotating and fixed wing models, it was deemed prudent also to perform a wind tunnel test on the 25 ft (7.6 m) flight-worthy rotor. This step results in considerable risk reduction in nonaerodynamic areas such as loads, dynamics, closed loop stability, etc. It also provides an excellent technology base for subsequent applications of the concept. The major ingredients of this phase are the detail design, fabrication, and tests of the blades, hub, and pneumatic system. Each of these aspects is discussed in the following sections.

5.3.1 Wing/Rotor Blade

The X-Wing rotor blade is actually designed more by the fixed wing criteria of aeroelastic divergence and blade loads than the familiar rotary constraints of fatigue and frequency placement. As a consequence, the planform resembles a high aspect ratio wing as shown in Figure 14 (but without the movable control surfaces). The structural configuration is very simple due to the symmetry of the airfoil and provides an unprecedented level of stiffness-to-weight ratio as suggested by the very high blade bending and torsion frequencies. Fabrication of the main spar is straightforward using the three box mandrel method shown in Figure 15.

Figure 16 shows typical tool-try box spars to which the leading and trailing edge 'blowing units' would be attached. The blowing units are comprised of a Coanda surface and upper slot lip assembly which is both bonded and mechanically attached as indicated in Figure 14. The slot lip is designed to flex open under pressure so that the (unblown) leading edge will form a smooth aerodynamic surface. A titanium root end fitting is bonded into the box spar lay-up with backup provided by mechanical fasteners.

Structural testing of the blade shown in Figure 17 has supported a very conservative design approach and suggests that substantial weight reduction well beyond the initial design target may be possible in future designs. Pneumatic and Coanda flow tests (Figure 18) have also been conducted.

5.3.2 Hub and Controls

Cyclic and collective air modulation to fore and aft blade ducts is accomplished using an eight segment gate valve, shown schematically in Figure 19. This nondynamic control system is functional in all flight modes and requires no special configuration changes from rotary to fixed wing flight. A 2-per-rev cyclic input can also be generated with the present scheme. Figure 20 shows the actual rotor valving system as fabricated and ready for insertion into the hub. The rotor hub (Figure 21) is designed for very high stiffness in flapping, inplane and torsion. Collective pitch for autorotation and test envelope exploration is also provided via large bearings and a high stiffness actuator system. A rotating aerodynamic fairing is fitted to the hub as indicated in Figure 13.

5.3.3 Test Module

The X-Wing rotor system is installed in the test module shown in Figure 22. The forward fuselage and tail boom are configured along the lines of the flight demonstrator preliminary design in order to properly account for the body-rotor aerodynamic interference. The module is heavily instrumented to provide extensive structural and aerodynamic data. Checkout of the module and certain performance, dynamic, and control power tests will be conducted at the Lockheed Rye Canyon facility (Figure 23), preparatory to entry into the NASA Ames 40 x 80 ft tunnel.

5.4 X-WING SUMMARY

A successful completion of Phase I, wind tunnel tests, will enable Phase II, flight vehicle tests, to be completed within two to three years. The technical potential of the concept is extremely promising with no problems of a fundamental or intrinsic nature identified to date. The primary performance goals of low rotor system weight and high cruise efficiency are being achieved by Lockheed with little difficulty. A recent study (Reference 19) has also indicated a major potential for low noise inherent in the CC rotor aerodynamics. At the present time the potential VTOL applications appear to encompass a very broad spectrum of military and commercial missions.

6. CONCLUSIONS

Laboratory Circulation Control (CC) research has encompassed over 4000 hours of wind tunnel occupancy and obtained more than 45,000 discrete tests points from two-dimensional airfoil and subscale rotor tests. From these experimental investigations and extensive

theoretical development, it has become increasingly evident that the CC aerodynamics has reached a point of technical maturity. Both the Circulation Control Rotor (CCR) program and the X-Wing Flight Demonstrator program appear to be fundamentally sound concepts offering wide applicability to both military and civilian markets.

The CCR program has been implemented as a viable advanced rotor concept. A 44 ft (13.4 m) diameter rotor has been designed by Kaman, all of the hardware manufactured, and the rotor system integrity verified in whirl stand tests and full scale wind tunnel tests. An XH2/CCR aircraft is presently being modified for flight tests early in 1979. Major technical attributes expected of the CCR are lower vibration levels, greatly improved maintainability and reliability, low noise levels, and reduced rotor torque. The CCR has potential applications to an advanced Navy helicopter, to the heavy lift helicopter, and a quiet "energy efficient" civil rotorcraft.

The status of the X-Wing program also has been presented. A high speed flight demonstrator has been designed by Lockheed and a 25 ft (7.6 m) diameter rotor and control system has been manufactured. Whirl tests have been completed. The full scale rotor is presently being tested in the helicopter mode, fixed wing mode, and conversion mode (100-percent to zero rpm and vice versa). Following a successful wind tunnel program a flight demonstrator is expected to be built and flown.

In conclusion, the circulation control rotor technology has exhibited considerable promise to date. Although flight demonstration is still forthcoming all present indications are that a major improvement in rotary wing VTOL capability does exist with this concept and that the combined resources of government and industry should be focused on these developments to facilitate their timely utilization.

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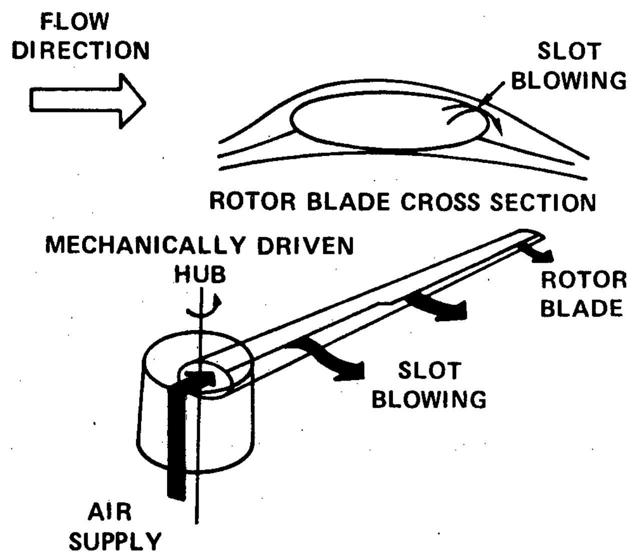


Figure 1 – Circulation Control Rotor - Basic Concept

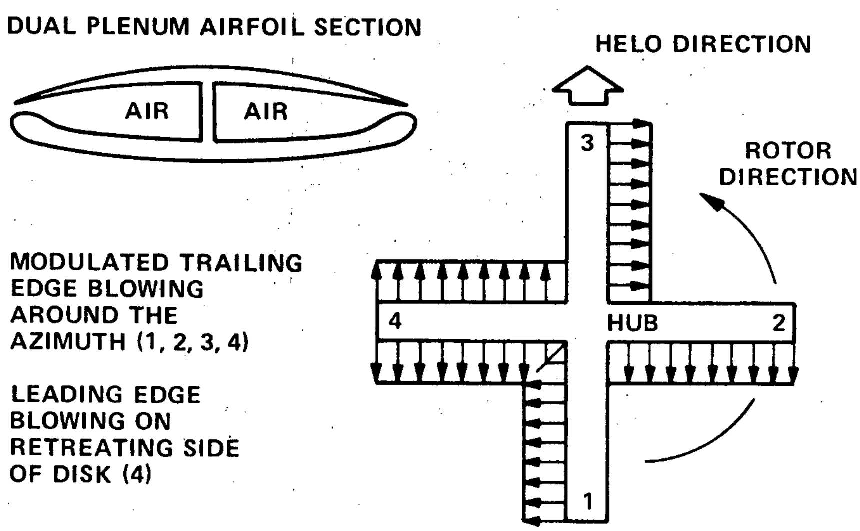


Figure 2 – Dual Blowing Concept for Operation at High Advance Ratios and during Conversion of the X-Wing

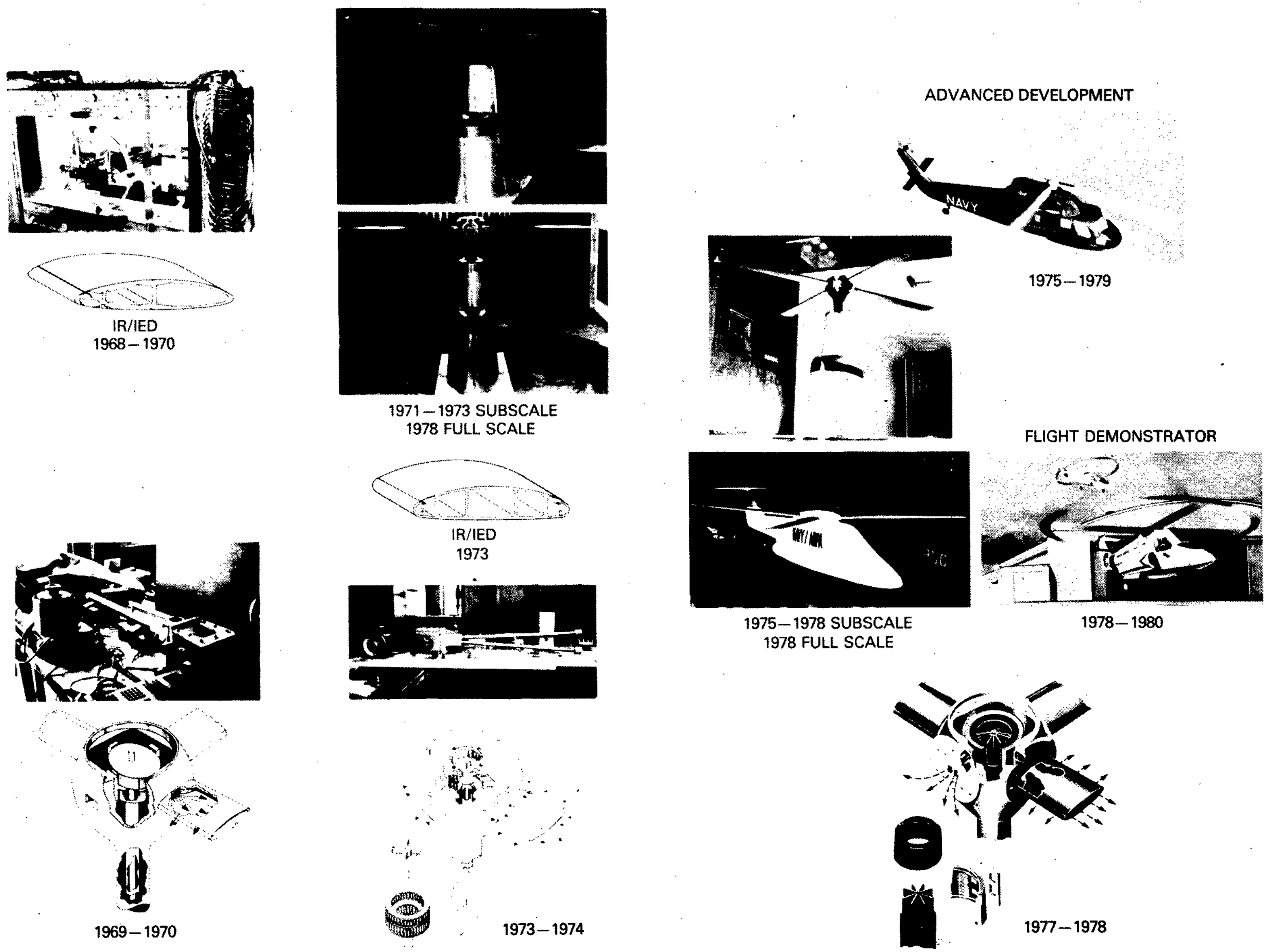


Figure 3 — Montage Showing the Chronological Development of Circulation Control

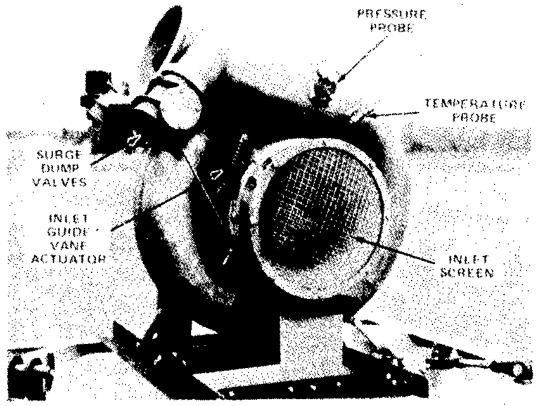


Figure 4 – XH2/CCR Compressor

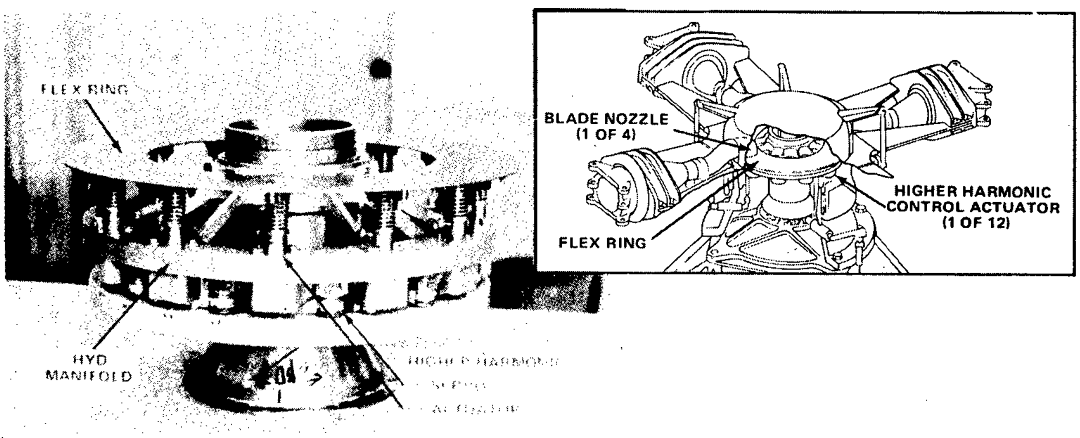


Figure 5 – XH2/CCR Nonrotating Hub Valve Assembly

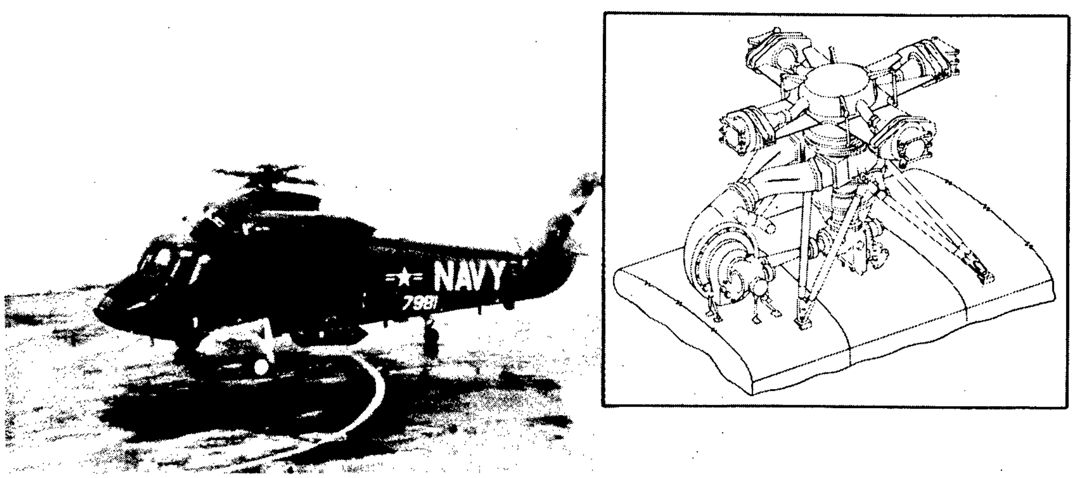


Figure 6 – XH2/CCR Test Bed Aircraft

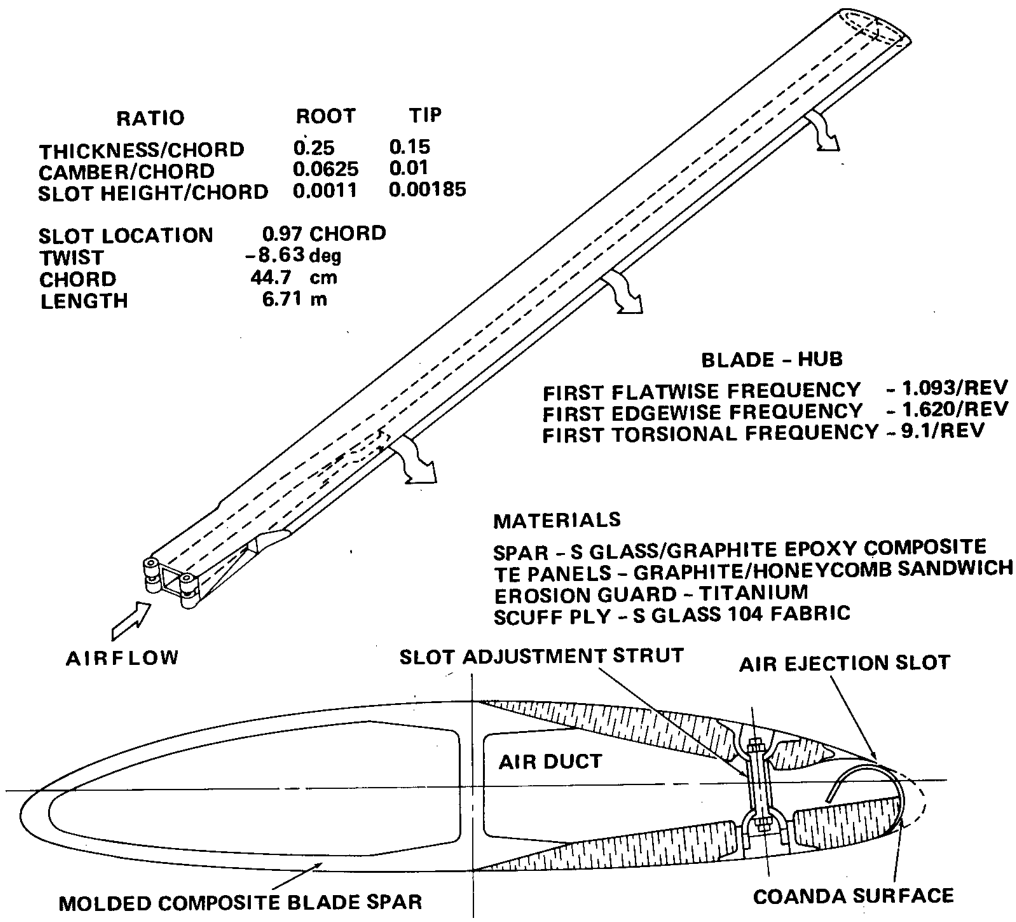


Figure 7 – XH2/CCR Blade Cross Section and Parameters

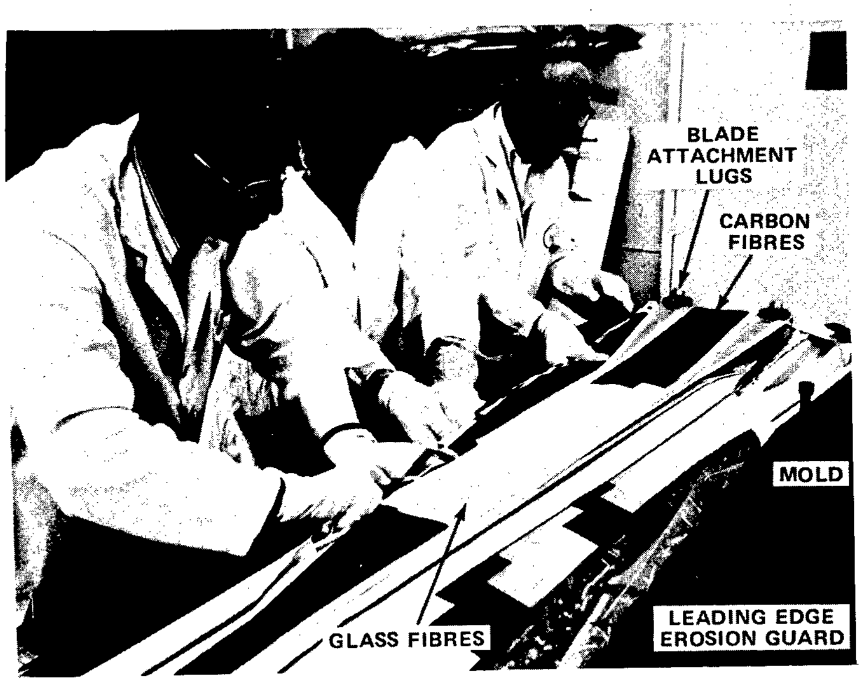


Figure 8 – Typical Blade Lay-up, Root Section

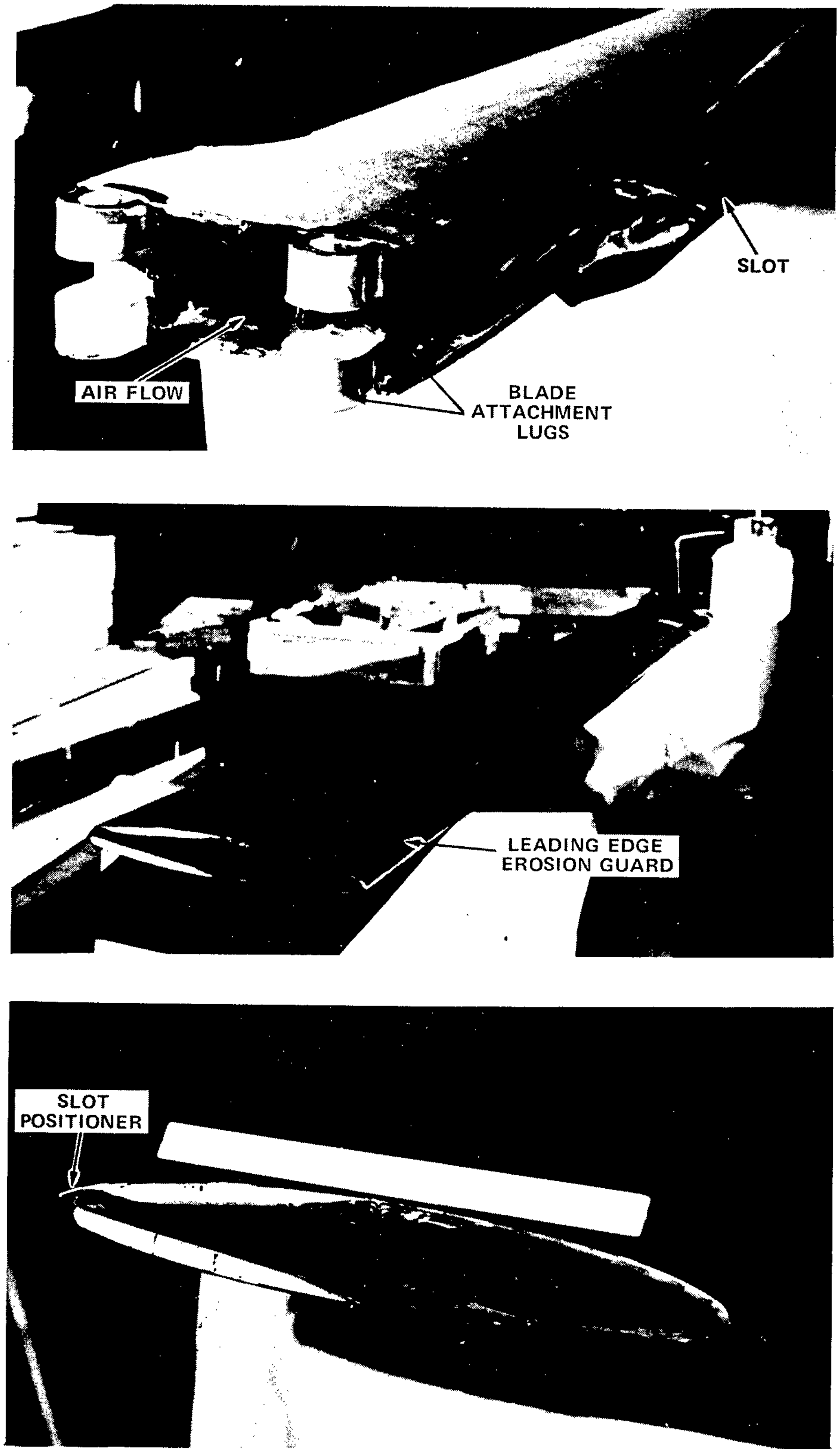


Figure 9 – XH2/CCR Blade

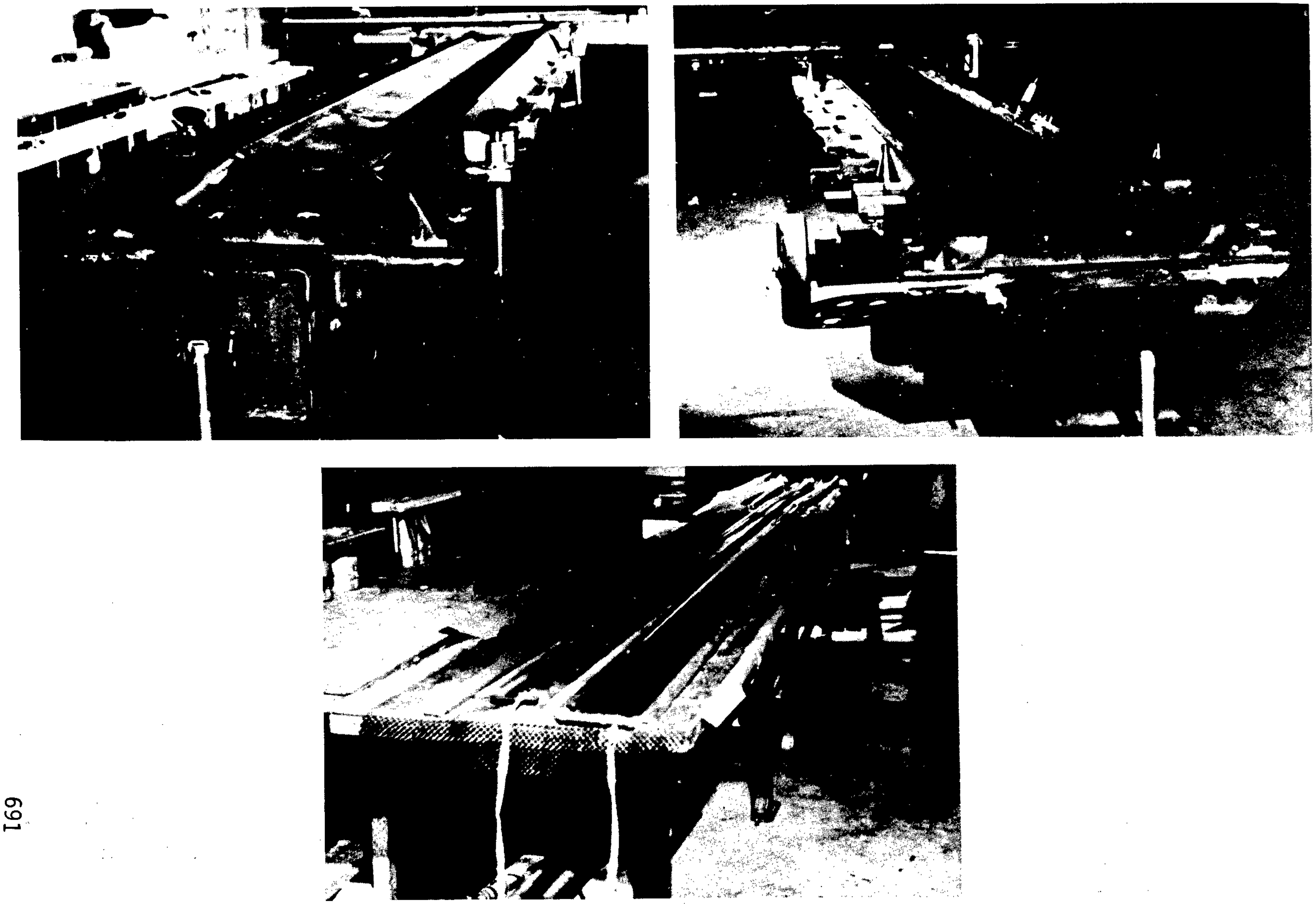


Figure 10 – XH2/CCR Blade Tooling

Figure 11 – XH2/CCR Rotor on KAC Whirl Stand

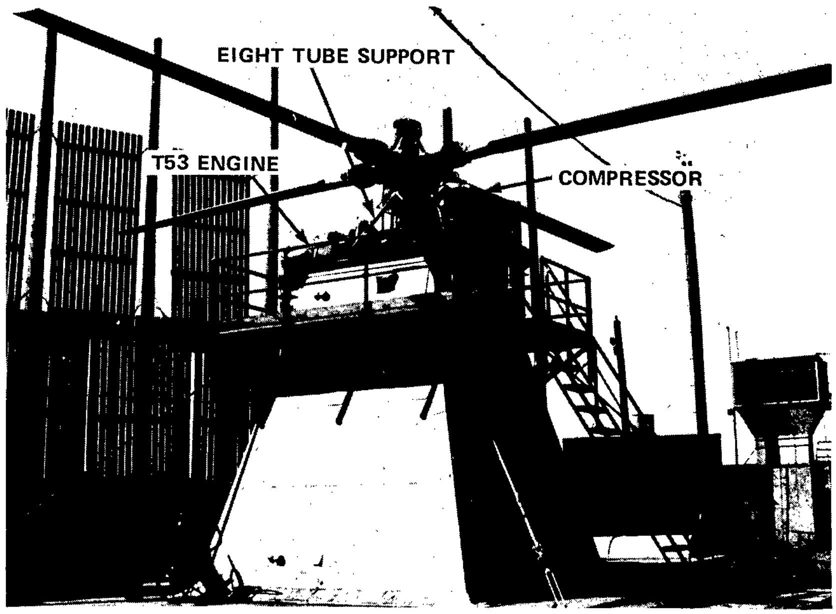
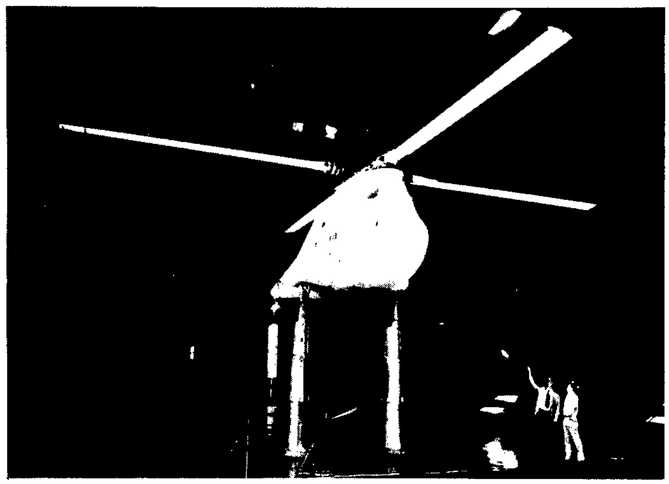


Figure 12 – CCR Installed in the NASA Ames Wind Tunnel



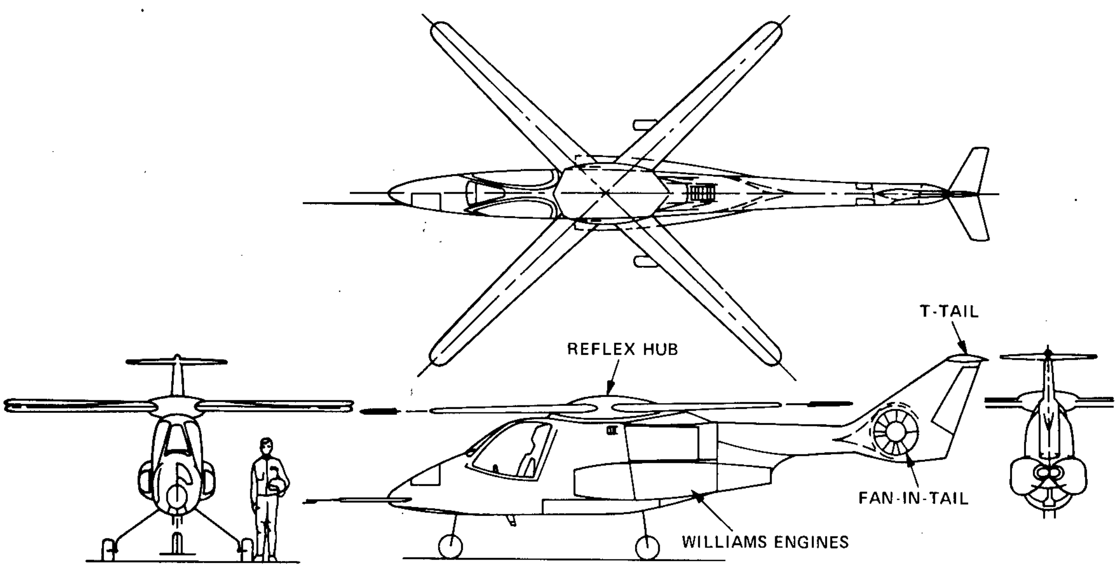


Figure 13 – General Arrangement of X-Wing Demonstrator

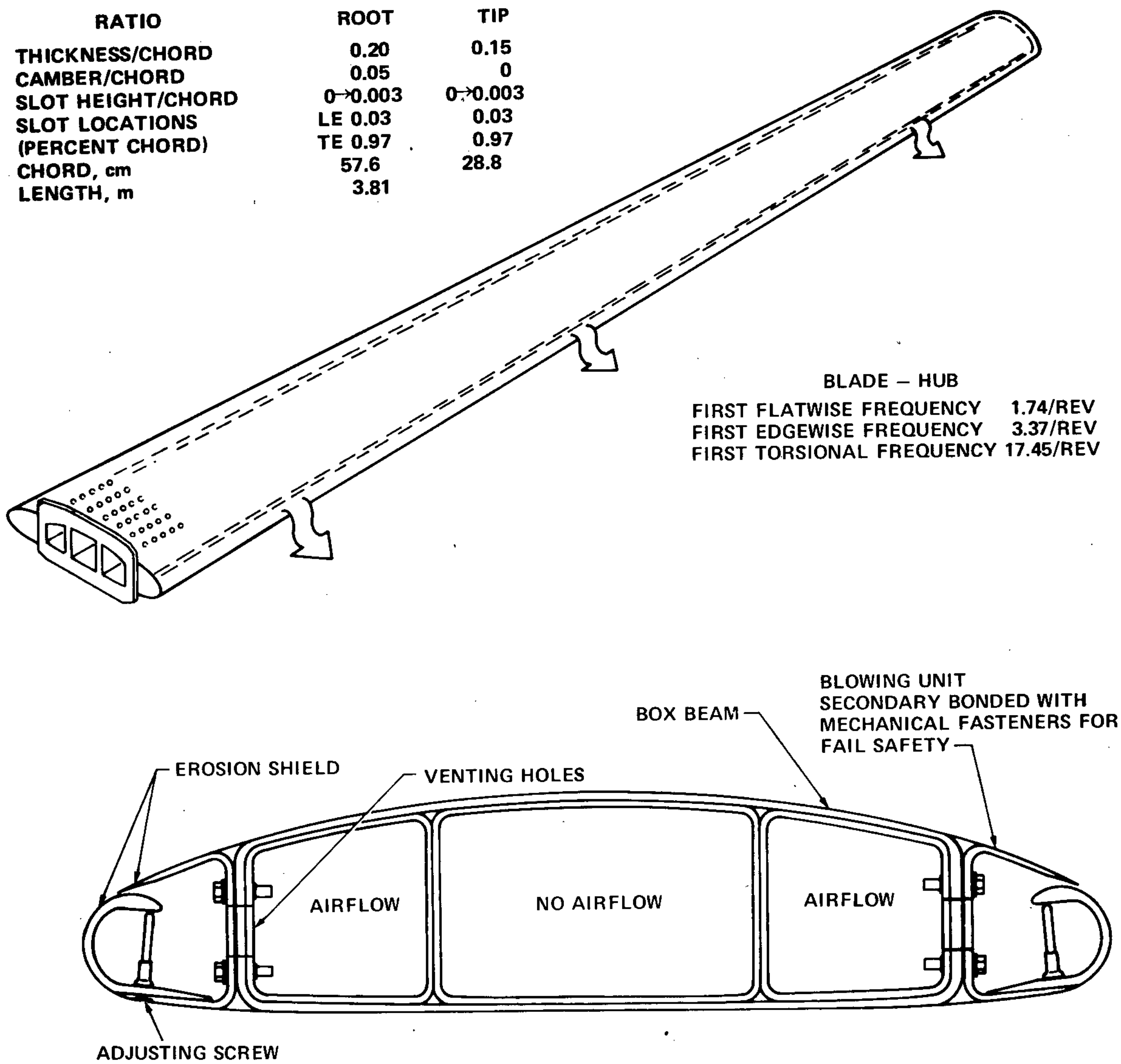


Figure 14 – X-Wing Blade Cross Section and Parameters

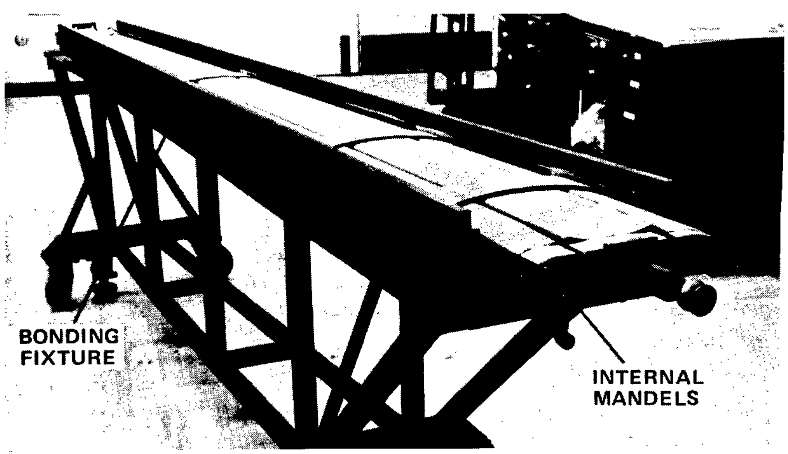


Figure 15 – X-Wing Box Beam Bonding Fixture

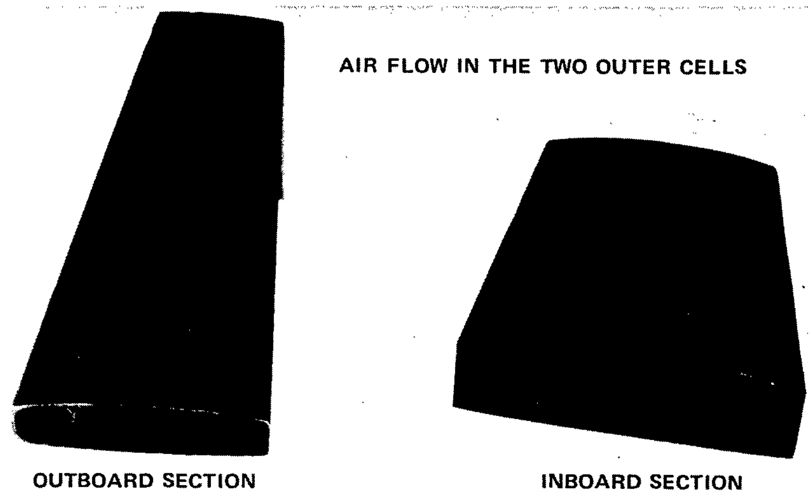


Figure 16 – X-Wing Tool Try Box Beam

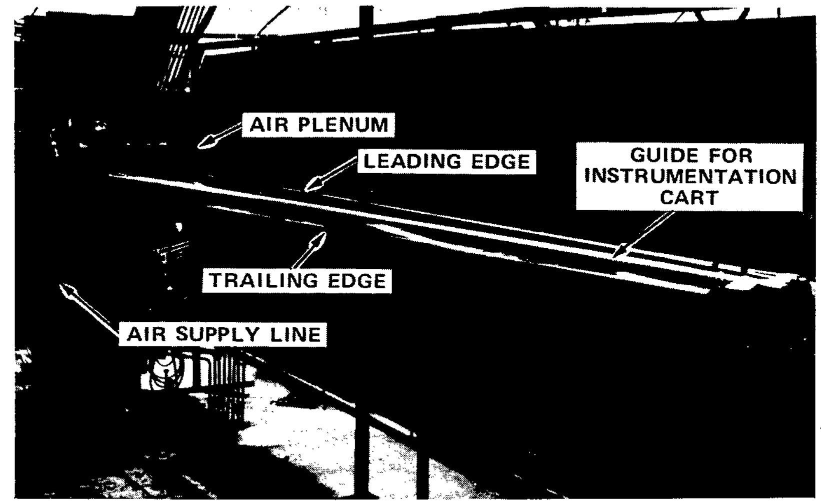


Figure 17 – X-Wing Blade w/o Tip Mounted for Static Testing

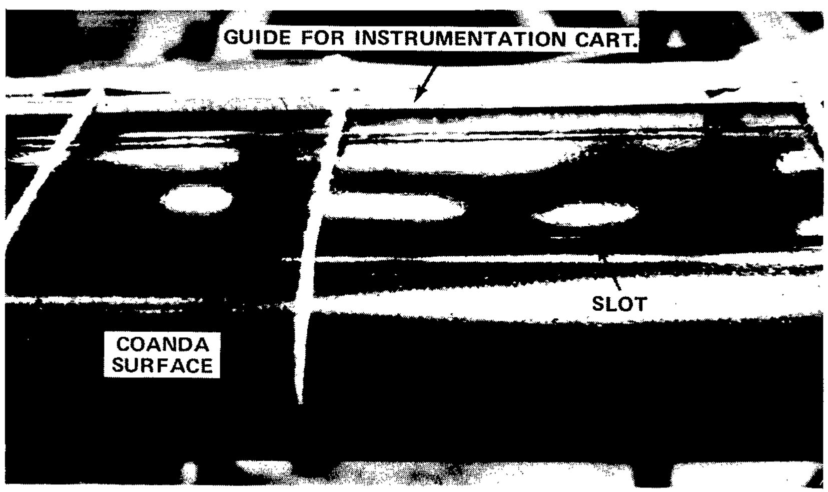


Figure 18 – Close-up Showing Slot Open, Duct Pressure = 574.6 N/m^2

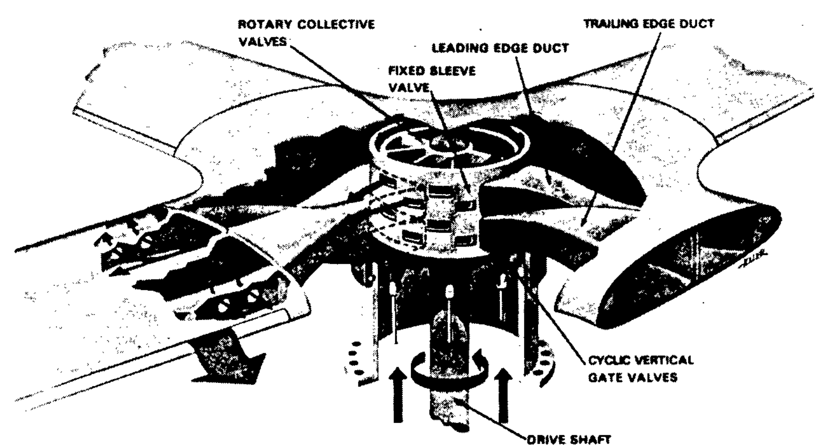


Figure 19 – X-Wing Hub and Control Concept

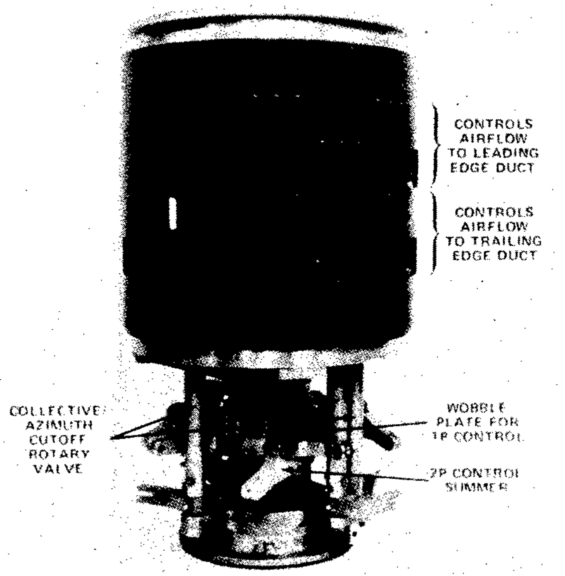


Figure 20 – Nonrotating Hub Valve Assembly

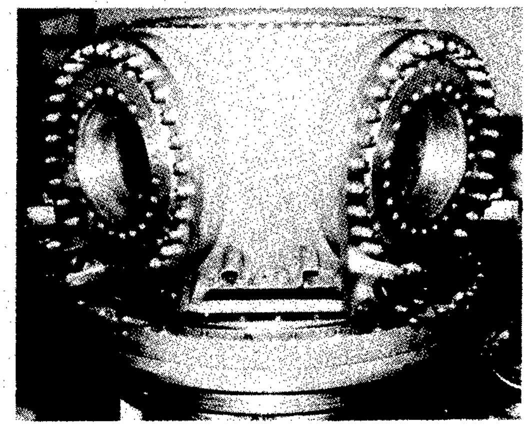


Figure 21 – Rotating Hub Structure

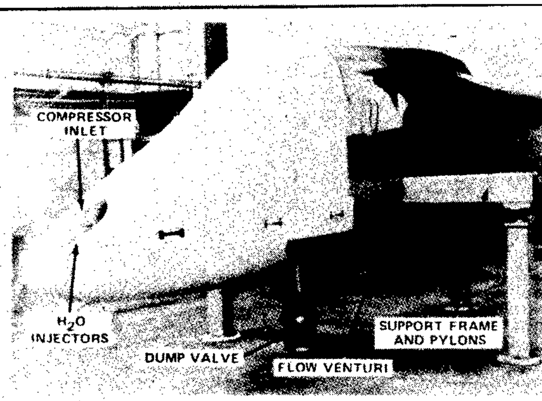


Figure 22 – Test Module for use in X-Wing 24 x 48 m Ames Wind Tunnel Test

Figure 23 – LAC Whirl Tower Used for X-Wing Testing

BIOGRAPHICAL SKETCH

Thomas M. Clancy was born in Brooklyn, New York, on May 15, 1926. He did undergraduate work in pre-engineering at Wesleyan University in Middletown, Conn., and received the Bachelor's degree in Aeronautical Engineering from Catholic University. He went on in 1950 to receive the Master's degree in Aero Engineering from Catholic U.

The early years of Mr. Clancy's engineering career were spent at the Naval Ordnance Laboratory where he developed missile configurations in the wind tunnels and ballistic firing ranges. Later he was an Aerodynamicist at the Martin Co. in Baltimore, Md., where for a time he was in charge of the performance of the Matador/Mace missile.

From 1960 to 1965, Mr. Clancy taught Mechancial Engineering at Howard University and also served as Aerodynamics Consultant to the Vitro Corporation and ACF Industries, Inc. The former involved aerodynamic performance and stability on the Tartar Missile and high altitude sounding rockets. Assignments at ACF involved aerodynamic design of a series of Navy and Air Force flight simulators.

In 1965, Mr. Clancy became a Project Engineer and later, Research Director at Booz, Allen Applied Research, Inc., in Bethesda, Maryland. While at Booz, Allen, he led a team of analysts in a study of the U.S. Supersonic Transport for the President of the United States. He also spent several years developing Navy/Marine rotary wing aircraft requirements for the Deputy Chief of Naval Operations for Air.

In 1971, Mr. Clancy began work for the Aviation and Surface Effects Department at DTNSRDC, specializing in rotary wing aircraft. He is presently Head of the Rotary Wing Group at DTNSRDC, -- the group which has been leading the development of the CCR helicopter and the X-Wing aircraft. For the past four years Mr. Clancy has also been an adjunct Professor of Engineering at Howard University in Washington, D.C.

AFFTC PARAMETER IDENTIFICATION EXPERIENCE

BY

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AFFTC Parameter Identification Experience

Abstract

One of the fundamental tasks of engineering and science, is the extraction of information from data. Parameter identification is a discipline that provides tools for the efficient use of data in the estimation of constants appearing in mathematical models of physical phenomena.

The application of parameter identification techniques to aircraft flight testing is simply the process of obtaining quantitative measures of various aircraft characteristics. In general, the parameters may relate to aerodynamic, structural, performance, or other types of characteristics. Typically, the flight-determined characteristics are compared with predicted values to verify or point out deficiencies in the predictions. They are used to substantiate design goals, to assess control system performance, to verify and improve piloted simulators, and to establish design criteria.

This paper presents an overview of AFFTC experience with parameter identification and presents the major results of such application with respect to accuracy, contributions of otherwise unobtainable information, cost effectiveness, and problems encountered.

Introduction

The Air Force Flight Test Center has been engaged in the determination of the coefficients of "model" equations (stability and control derivatives) which describe the flight characteristics of air vehicles since 1948. In addition, from 1948 to 1961, no less than twenty-two publications on the subject were authorized by the National Aeronautics and Space Agency (NASA, then NACA), the United States Air Force and other agencies (reference 1).

The advantages and benefits of developing a complete mathematical description of an air vehicle's stability and control characteristics has been long recognized by the flight test community. They include (reference 2):

1. Improved verification of performance criteria,

2. Improved dependability of extrapolated flight characteristics,

3. Enhanced system development and optimization of vehicle performance,

4. More representative engineering and operational simulators, and

5. The reduction of the amount of flight test time required to adequately assess the flight characteristics of an air vehicle.

Despite the recognition of the desirability of accomplishing such a task, early parameter identification efforts were limited to the synthesizing of major stability derivatives from data produced by small, linear control inputs. The manual matching of analog computer outputs with flight test traces (analog matching), using linear equations of motion was the most commonly used method of determining these major derivatives (reference 3). Although good results were (and are) achieved, this technique was time consuming and was strongly dependent on the sophistication of the operator. The development of faster, more accurate methods has been accomplished in recent years using improved mathematical techniques

well suited for automated computer application. The remainder of this paper concerns the use and application of an automated parameter identification scheme as applied to flight test data at the AFFTC.

The particular parameter identification technique which has been widely used at the AFFTC is the Modified Maximum Likelihood Estimator (MMLE) developed by Kenneth W. Iliff and Lawrence W. Taylor, Jr. of the NASA Dryden Flight Research Center, Edwards AFB, CA. Experience with this method has been restricted to identification of parameters of linear mathematical models.

Over the past several years, this method has been applied on ten major test programs; X-24B, YF-16, YF-17, A-9, A-10, YC-14, YC-15, F-15, F-16, and B-1. The uniqueness of application at the AFFTC has been in its use as the first application of parameter indentification as a production analysis tool. The AFFTC has processed more than 1500 maneuvers in limited amounts of time using parameter identification techniques.

Parameter Identification Experience

The current philosophy of evaluating aircraft at the AFFTC directs aircraft testing efforts toward three major areas; system development, compliance with performance criteria, and minimum testing required. Use of parameter identification techniques has yielded significant improvements toward meeting these goals over previous analysis techniques employed. Due to the nature of this technique, identification of detailed mathematical models is accomplished, thus increasing the information available by which the overall system can be more effectively analyzed and developed.

Test Optimization:

The aircraft flight test maneuvers required for parameter identification can be of a different type than is used to obtain the more classical test data and thereby provide an independent test whose results can be directly correlated with classical testing. The independence of test techniques also allows for an optimum test plan to be developed which utilizes both the new and more classical test methods, thereby minimizing the

overall flight test time required. From an implementation point of view, we feel that we have demonstrated significant reductions in the amount of flight test time necessary to define the stability and control characteristics of an air vehicle. During the evaluation of several prototype and production aircraft, records were maintained with respect to the amount of dedicated flight time devoted to classical stability and control maneuvers and those flight hours devoted to parameter identification of STABility Derivative EXtraction (STABDEX) maneuvers.

Table I

A COMPARISON OF CLASSICAL AND STABDEX FLIGHT REQUIREMENTS FOR A TYPICAL CONFIGURATION

	Classical	STABDEX
Total Maneuvers	159(89)	117(53)
Total Flight Hours	20.8(11.7)	4.2(1.9)
Maneuvers per Flight Hour	7.6(7.6)	27.9(27.9)
Parameters per Maneuver	2.9(2.9)	10.5(10.5)
Parameters per Flight Hour	22(22)	293(293)

NOTE: Numbers in parenthese are for an idealized flight test program.

Table I is a listing of the actual experience during one of these programs in which both classical and STABDEX methods were used. The numbers in parentheses are an estimate of an idealized test program, based on hindsight, which could have been flown to obtain the same data. Two points are obvious. The first is that efficient test planning can effect a significant reduction in flight test time regardless of the analysis method used. The second is that the total flight time can be reduced nearly 75% by the application of STABDEX techniques. It must be noted that the flight time indicated in the tables is not exactly representative of the total flight time required since the

evaluation of characteristics such as the variation of pitch control force and deflection with velocity and certain roll performance parameters must be obtained by classical techniques. On the other hand, in a properly implemented active control system, the longitudinal short period frequency and damping ratios can only be quantified by STABDEX techniques.

Accuracy:

The most fundamental and commonly asked question concerning parameter identification techniques is; how accurate are the results? The term accuracy implies an absolute measure of the error between an estimate and the true value. This of course is impossible to compute since the true value is unknown; however, there are several indicators that have been used which lend confidence and credibility to the results obtained by the application of parameter identification to test data. The three basic indicators of accuracy in the results are: (1) repeatability, (2) correlation, and (3) statistical error analyses.

Figures 1 through 4 shows data which were obtained from an aircraft using the MMLE computer program. Each data point plotted vs angle of attach (α), represents an independent test condition. The data exhibits significant repeatability where data were obtained at similar test conditions. Since these tests were independently conducted and evaluated, the repeatability exhibited lends confidence that the technique yields consistent results. The data presented here is typical of the results obtained on many other test programs conducted at the AFFTC. The vertical lines presented on these plots represent "confidence" levels as computed by the MMLE program. These "confidence" levels will be discussed later.

Figures 5 and 6 are typical examples of the correlation obtained between classical and STABDEX techniques. The data symbols plotted on figure 5 were measurements taken directly from steady-heading sideslip maneuvers, while the faired lines depict data obtained through calculation using STABDEX data. Figure 6 shows similar data; however, this data was obtained over a wide range of flight conditions and is graphic evidence that

STABDEX technique yields nearly identical results as does the classical steady-heading sideslip maneuvers.

A third technique which has been used to validate STABDEX data is the comparison of in flight measured frequency response estimates; that is, measured estimates of the aircraft transfer function, with the Laplace transformation of the equations of motion, where the Laplace transforms are computed using STABDEX data. The result is a direct comparison between measured and computed aircraft transfer functions in the frequency domain. With respect to the dominant model parameters, the results of this technique showed the method to be sensitive enough to easily verify the STABDEX derived model.

In the computational scheme employed in the MMLE program, there exists the capability (under certain conditions and restrictions) to calculate the statistical variance of the estimated parameter value with respect to the true value. This has been shown to be easily accomplished provided one also obtains a measure of the noise associated with the inflight measured variables. If this can be accomplished, a correction may be made to the "confidence" level which is computed by the MMLE program and which results in an estimate of the variance. The correction factor to be multiplied is N/2B where N is the sample rate and B is the measured frequency bandwidth of the noise (reference 4). This technique has not been employed to data in a production test program; however, we feel that application of this technique will become standard practice. It should be noted that the "confidence" level which is calculated has been used as a relative measure of goodness and parameter sensitivity as can be seen from figures 1 through 4.

Flight Test Parameter Identification Comparisons with Wind Tunnel:

Prior to first flight, wind tunnel estimates remain our primary indicators of how an aircraft will behave. However, there have been several instances where wind tunnel estimates failed to adequately predict aircraft response particularly where initial placard limits were to be established. It is not surprising that differences between wind tunnel estimates and flight test data occur in the rotary derivatives, but, the area of concern, in

my opinion, is in the variation experienced in the major derivatives. Figures 7 and 8 are examples of the differences experienced between wind tunnel estimates and flight test data for two major derivatives. Figure 7 shows the wind tunnel estimate for the static margin parameter (C_{m_α}) and shows zero static margin actually occured near 22.5 degrees angle of attack as opposed to the predicted 14.5 degree angle of attack crossover point. Figure 8 shows the directional stability parameter (C_{n_β}) to be much more stabilizing than predicted. These data have been verified by the techniques mentioned earlier. This discussion is not intended as a condemnation of wind tunnel data or methods, but, rather serves to point out the need for flight testing and increased communication between the flight test and wind tunnel communities.

Parameter Identification and Configuration Effects:

One of the most powerful results of the STABDEX method is in the ability to model relatively small changes in the aircraft's response characteristics due to changes in the aircraft's shape or configuration. Figures 9 and 10 show data obtained for one prototype aircraft. Data shown in figure 9 represents a comparison of static lateral-directional stability characteristics with a change in external stores loading (configuration effect), while the data in figure 10 depicts a change in static lateral - directional stability characteristics with dynamic pressure variations (flexibility effects). It has been our experience that as long as any physical change in the aircraft occurs and produces a measurable change in aircraft response, application of the STABDEX technique has yielded accurate mathematical models of the phenomena.

Other Contributions of Parameter Identification

The examples I have given are only a few of the many which have been encountered in recent years. There are many contributions which have been made through the use of parameter identification which have enhanced our ability to analyze, evaluate, and optimize aircraft performance. One of the advantages in developing a mathematical model of the aircraft is the ability to extrapolate a measured parameter to the next most hazardous

flight condition, thereby enhancing safety and mission effectiveness. This has been practiced on more than one test program with great success, particularly near the extremes of the flight envelope. On one prototype test program of a multiengine aircraft, data was obtained during engine-out approaches to stalling maneuvers while maintaining wings level and zero sideslip. The raw measurements of aileron deflection were corrected for aileron required due to rudder deflection yielding the aileron requirement due to an engine-out rolling moment. The measured data were corrected by application of the inflight determined roll control effectiveness parameters (control derivatives) $C_{\ell_{\delta_r}}$ and $C_{\ell_{\delta_a}}$. Figure 11 illustrates the results of those calculations for one configuration tested. The data presented shows how the STABDEX data was utilized to compute and extrapolate the aileron requirements for an engine-out situation to the worst case flight conditions, thereby establishing a margin of safety which would have been difficult or impractical to establish otherwise.

Many evaluations which have been recently accomplished would have been impossible to conduct without the use of parameter identification. This is becoming more and more commonplace with the trend toward active control systems. The two most obvious of all contributions of parameter identification are in the evaluation and optimization of the flight control systems associated with the aircraft, and our ability to provide accurate mathematical models for engineering or operational training simulators.

Concluding Remarks

The flight test community at the AFFTC has had significant success with the method of parameter identification. Our successes have included the reduction and optimization of flight test programs, improvement in our ability to verify performance criteria, enhanced system development and optimization of vehicle performance, and improvement in the dependability of measured flight characteristics which has also led to more accurately represented simulations. We are committed to the continued use and development of parameter identification techniques, and expect further improvement in flight testing

will occur with the development of non-linear model identification programs and broader applications.

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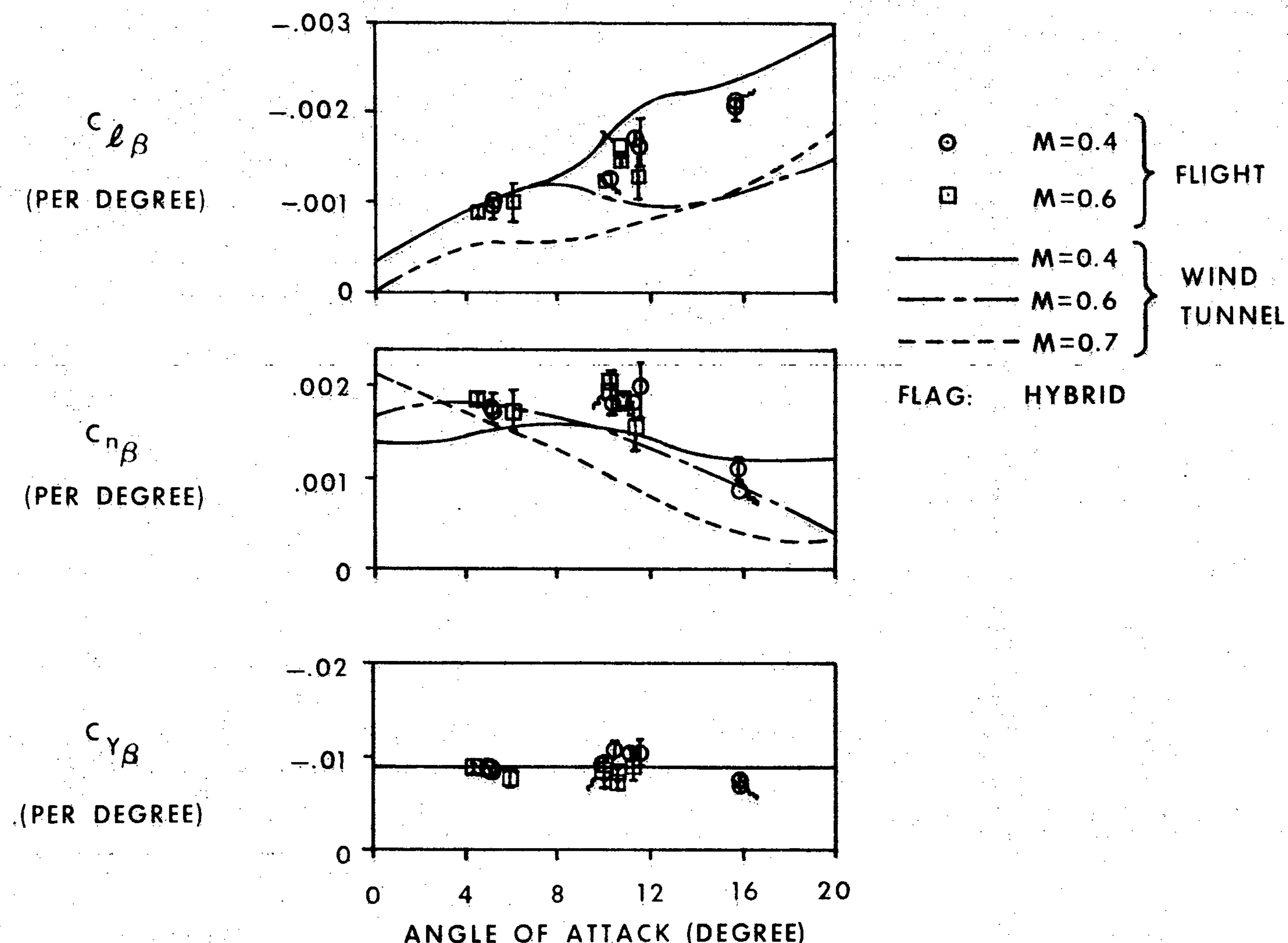
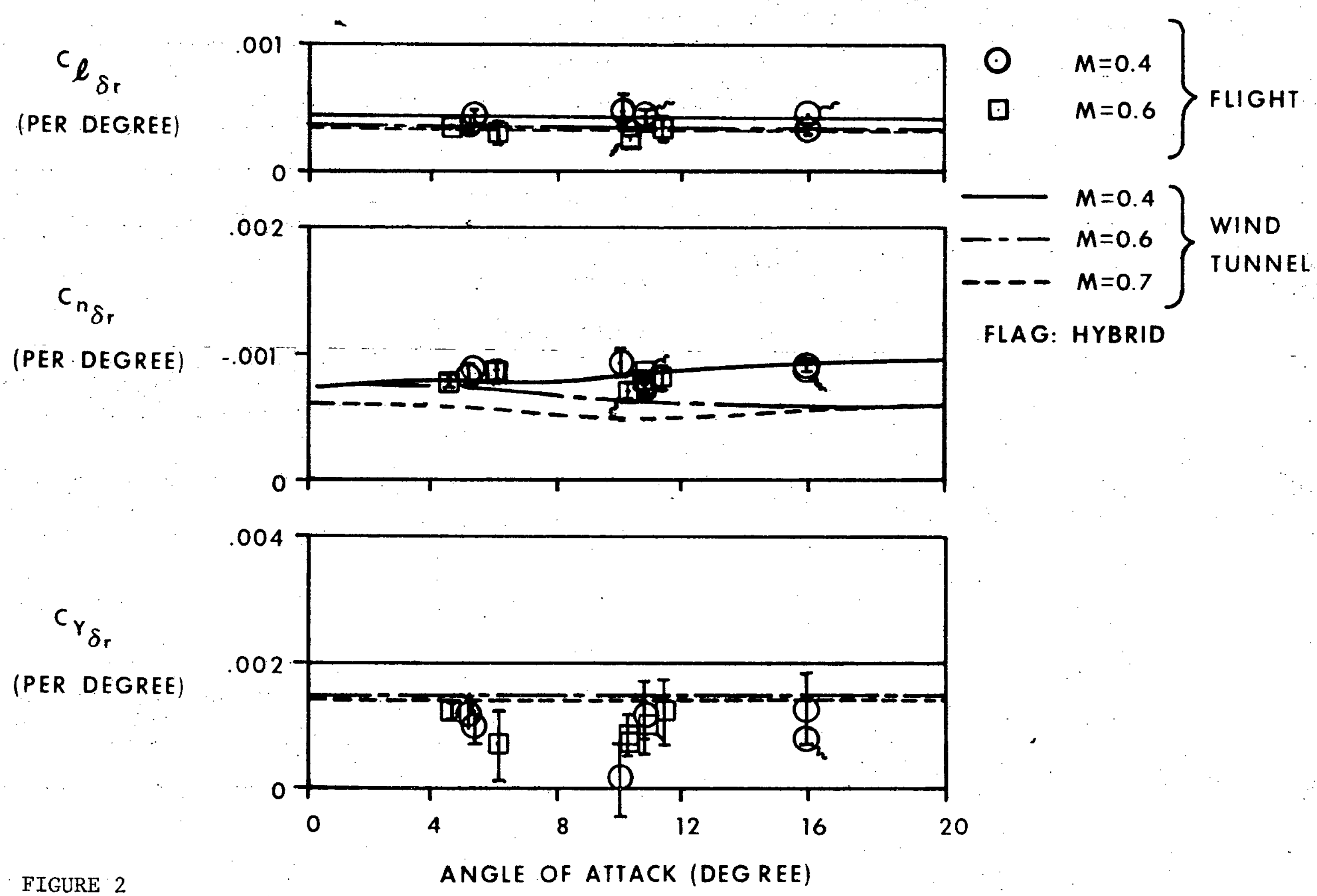


C_{ℓ_β}
(PER DEGREE)
−.003
−.002
−.001
0
C_{n_β}
(PER DEGREE)
.002
.001
0
C_{Y_β}
(PER DEGREE)
−.02
−.01
0
0 4 8 12 16 20
ANGLE OF ATTACK (DEGREE)
M=0.4
M=0.6
FLIGHT
M=0.4
M=0.6
M=0.7
WIND TUNNEL
FLAG: HYBRID

FIGURE 1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$C_{n_{\delta r}}$
(PER DEGREE)
$C_{Y_{\delta r}}$
(PER DEGREE)
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.002
-.001
0
.004
.002
0
0
4
8
12
16
20
ANGLE OF ATTACK (DEGREE)
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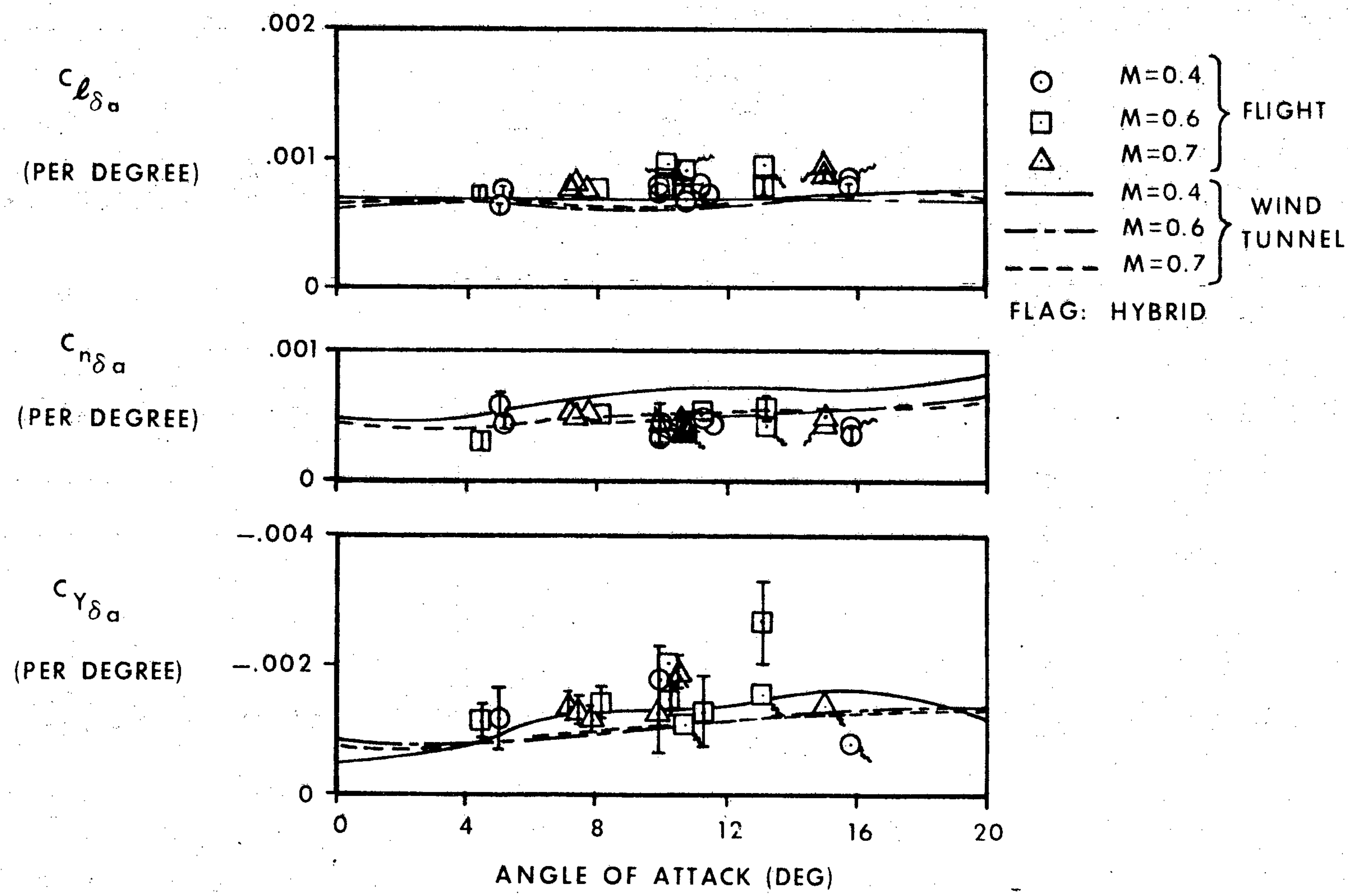
FLIGHT
M=0.4
M=0.6
M=0.7
WIND TUNNEL
FLAG: HYBRID

FIGURE 2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$C_{n_{\delta_a}}$
(PER DEGREE)
$C_{Y_{\delta_a}}$
(PER DEGREE)
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0
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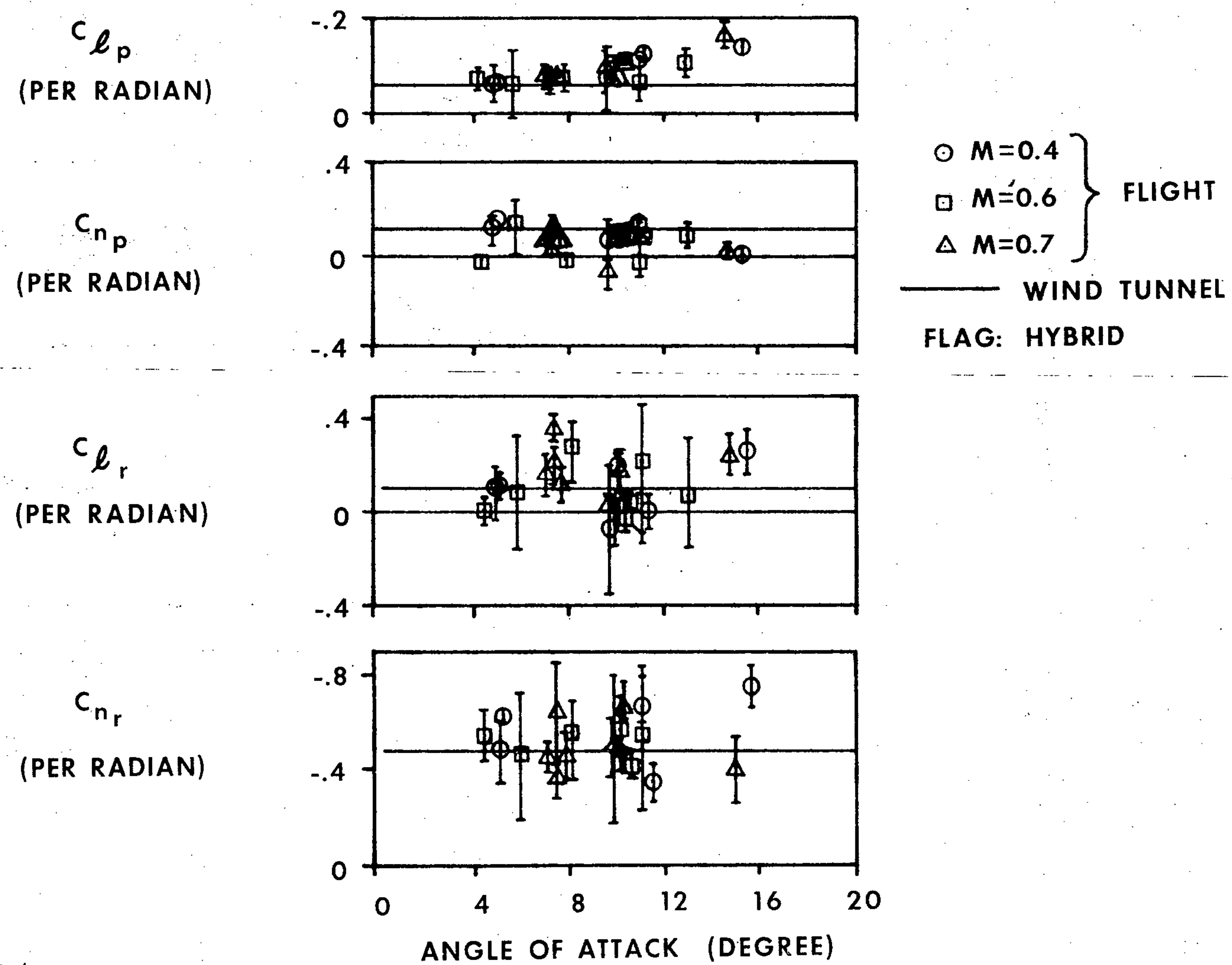
−.002
0
0
4
8
12
16
20
ANGLE OF ATTACK (DEG)
M=0.4
M=0.6
M=0.7
FLIGHT
M=0.4
M=0.6
M=0.7
WIND TUNNEL
FLAG: HYBRID

FIGURE 3


C_{ℓ_p}
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C_{n_p}
(PER RADIAN)
C_{ℓ_r}
(PER RADIAN)
C_{n_r}
(PER RADIAN)
-.2
0
.4
0
-.4
.4
0
-.4
-.8
-.4
0
0
4
8
12
16
20
ANGLE OF ATTACK (DEGREE)
M=0.4
M=0.6
M=0.7
FLIGHT
WIND TUNNEL
FLAG: HYBRID

FIGURE 4

STEADY HEADING SIDESLIP CHARACTERISTICS

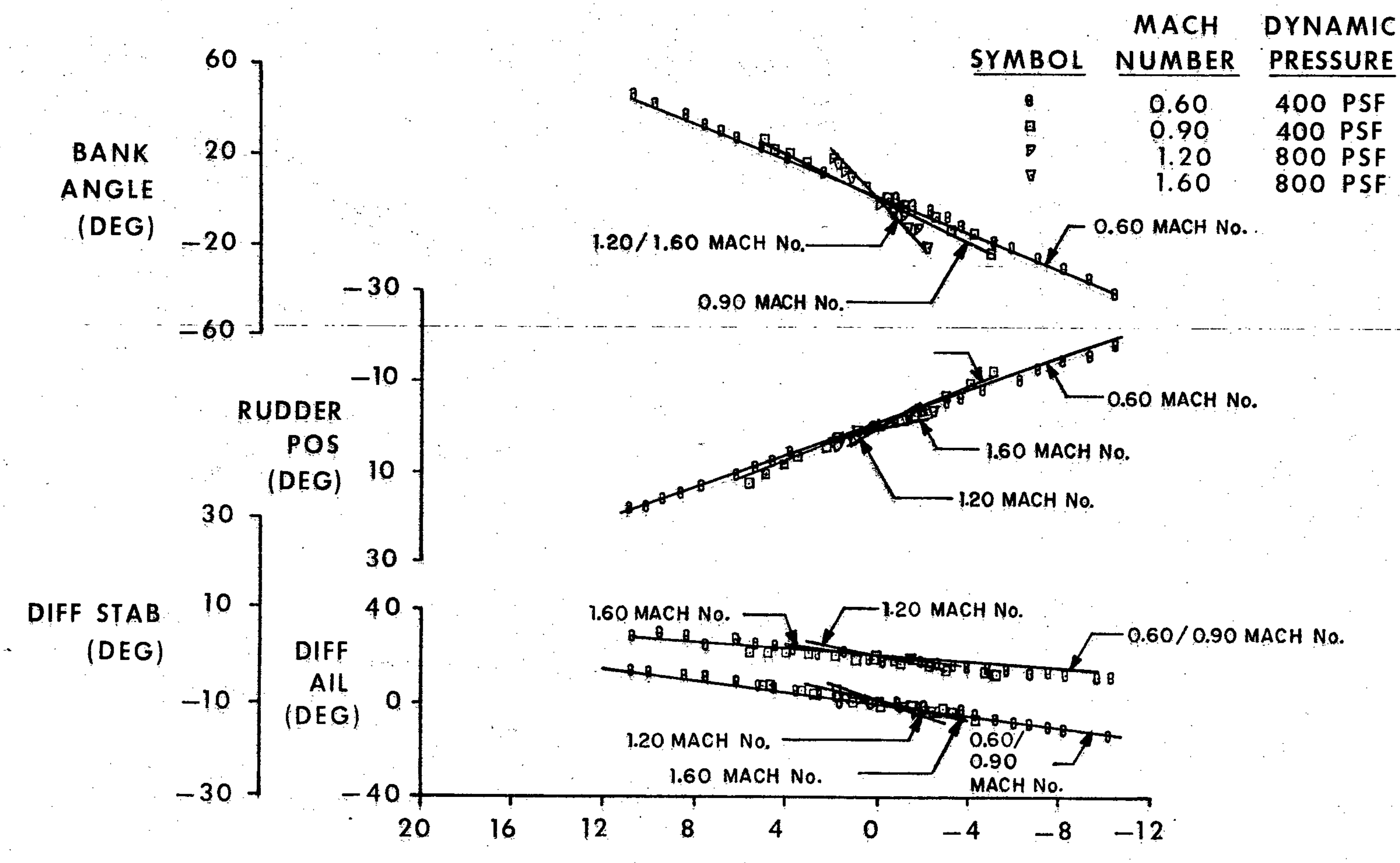
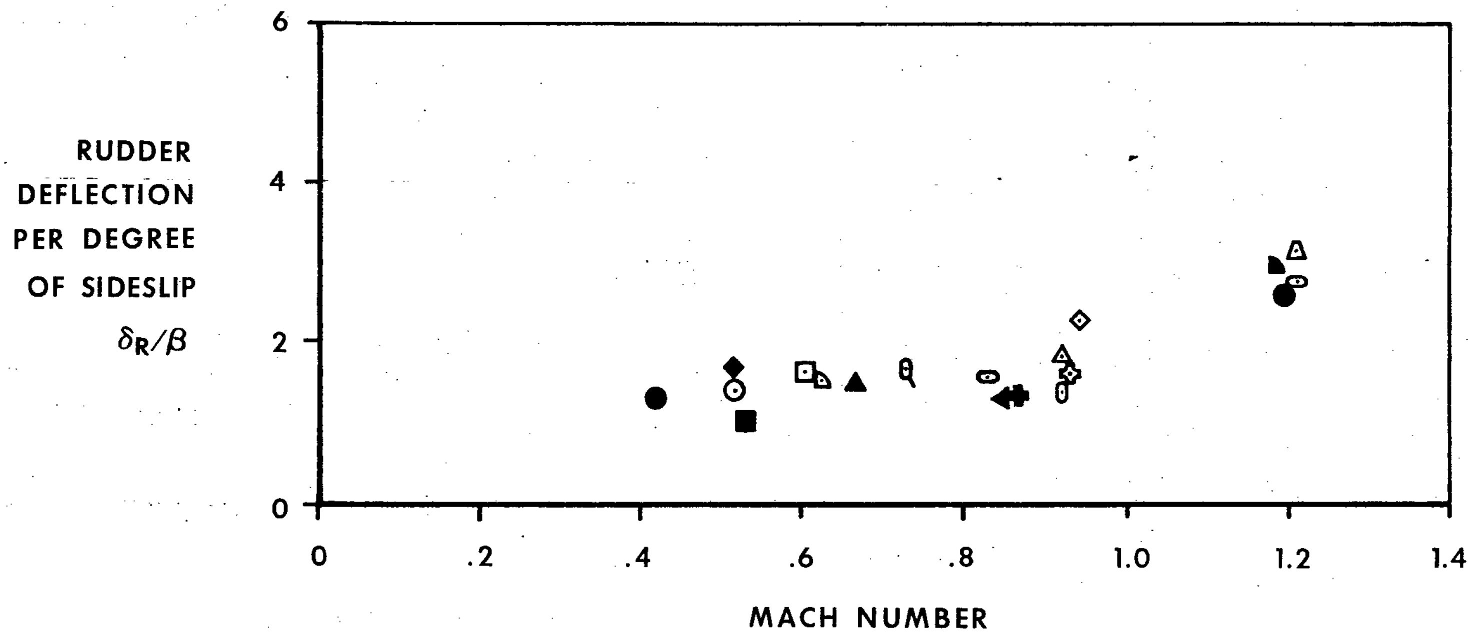


FIGURE 5 NOTES: 1. Data points were obtained from steady-heading sideslip maneuvers.
2. Fairings depict slopes defined by stability derivative data.

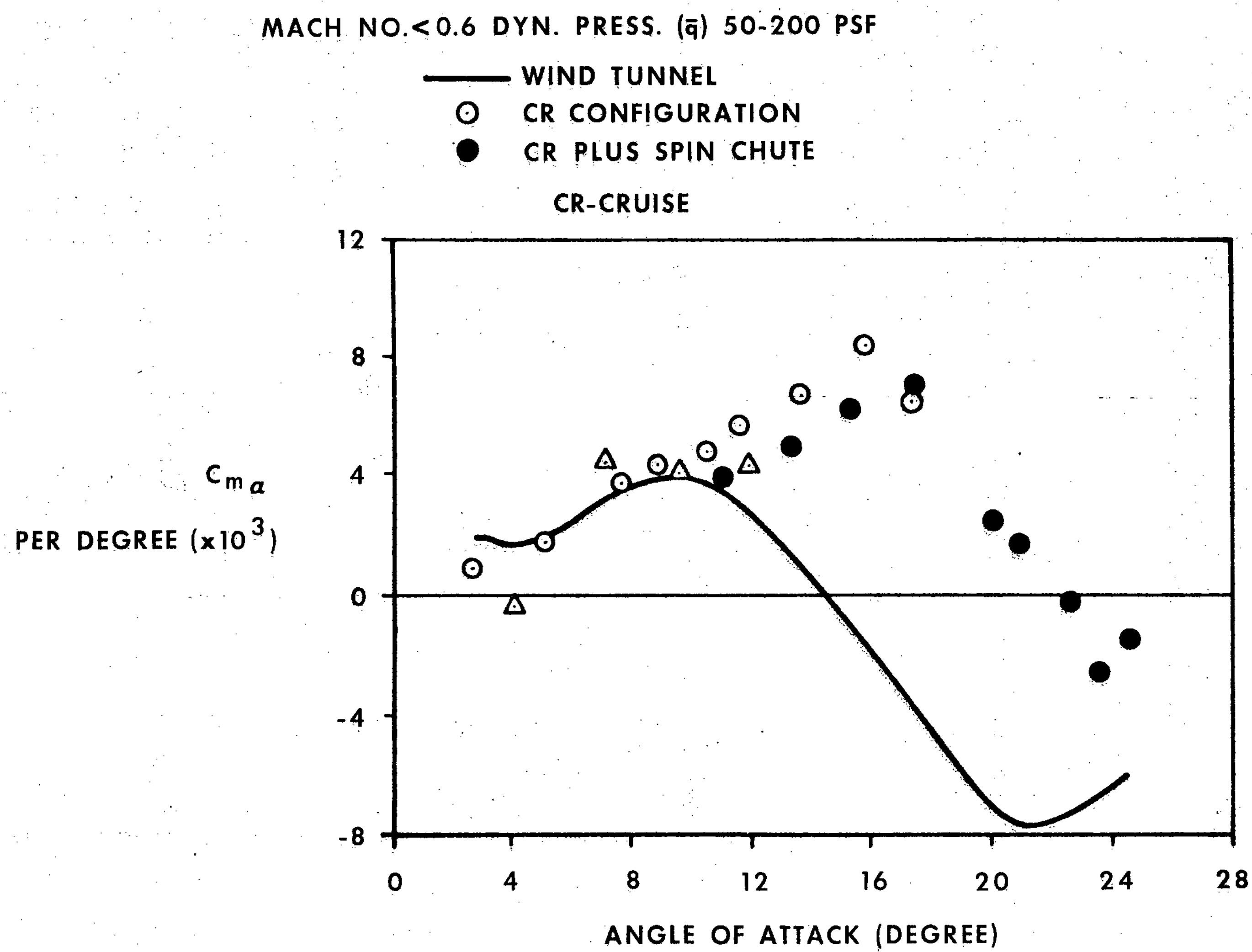
STATIC DIRECTIONAL STABILITY



NOTES: 1. Solid Symbols denote $\delta_r/_\beta$ calculated from inflight determined stability derivatives.

2. Open symbols denote $\delta_r/_\beta$ obtained from wings level and constant heading sideslips.

FIGURE 6



MACH NO.< 0.6 DYN. PRESS. ($\bar{q}$) 50-200 PSF
WIND TUNNEL
CR CONFIGURATION
CR PLUS SPIN CHUTE
CR-CRUISE
C_{m_α}
PER DEGREE ($\times 10^3$)
12
8
4
0
-4
-8
0
4
8
12
16
20
24
28
ANGLE OF ATTACK (DEGREE)

FIGURE 7

NOTES:

1. Test data are in the body axis, wind tunnel data are in the stability axis.
2. Fairings from wind tunnel estimates.

SYMBOL	NOMINAL THRUST COEFFICIENT (C_μ)
⊙	0.57
⊡	0.80
△	0.91
◇	1.00
◿	1.36
△	1.55

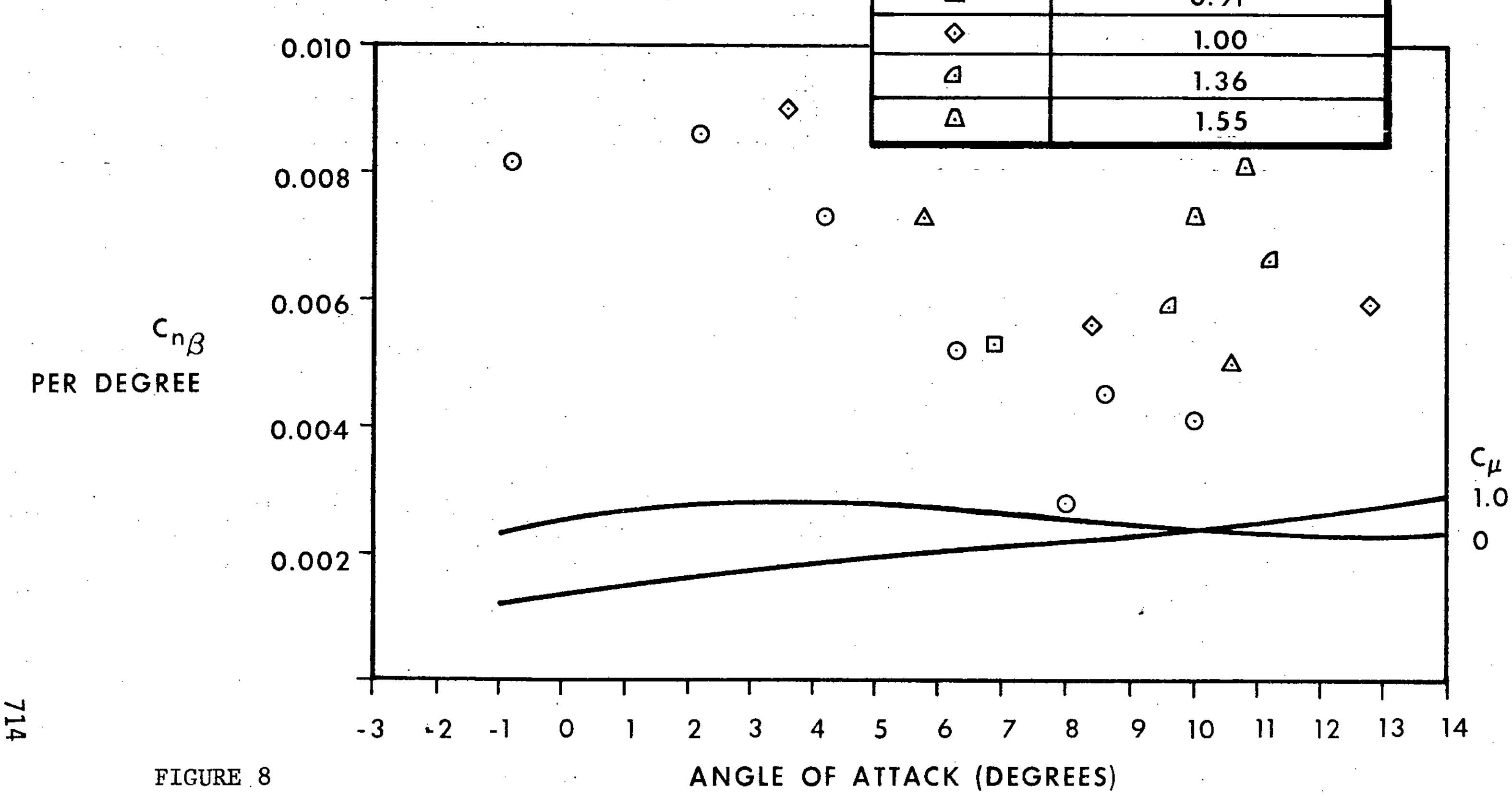


FIGURE 8

STATIC LATERAL-DIRECTIONAL STABILITY COMPARISON

LOADING 3

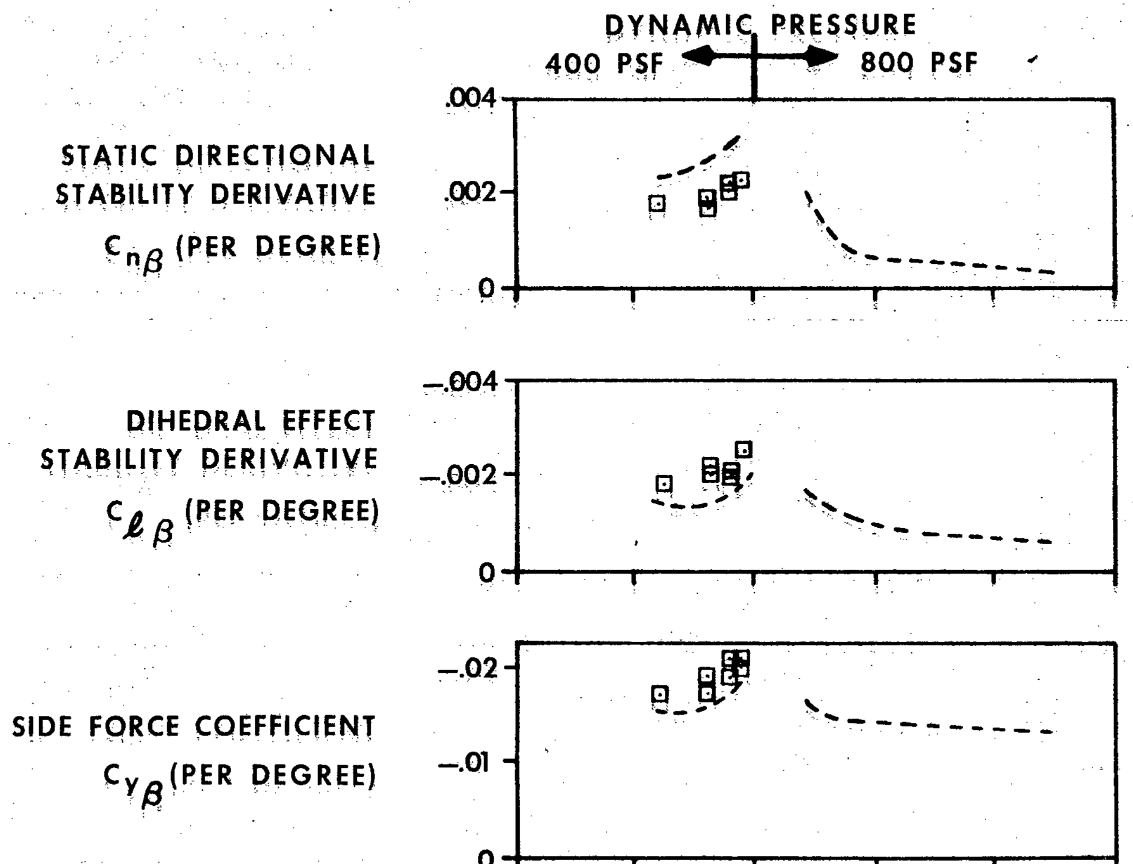
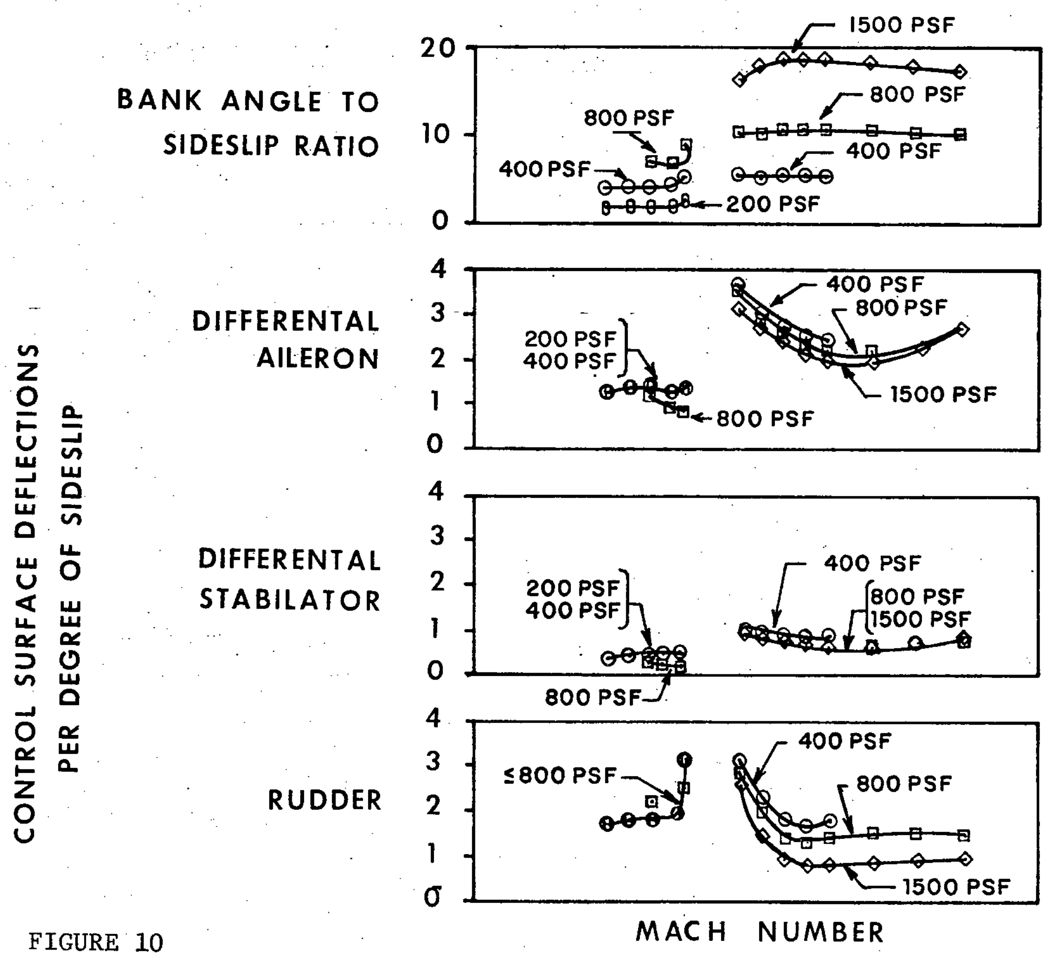


FIGURE 9 NOTE: Dashed lines represent loading 1.

STATIC LATERAL-DIRECTIONAL STABILITY



SYMBOL	DYNAMIC PRESSURE
⊖	200 PSF
⊙	400 PSF
⊡	800 PSF
◇	1500 PSF

NOTES:

1. Data points were computed from stability derivative data.
2. Fairings represent data point fairings

FIGURE 10

ASYMMETRIC THRUST CONTROL REQUIREMENTS - AILERON

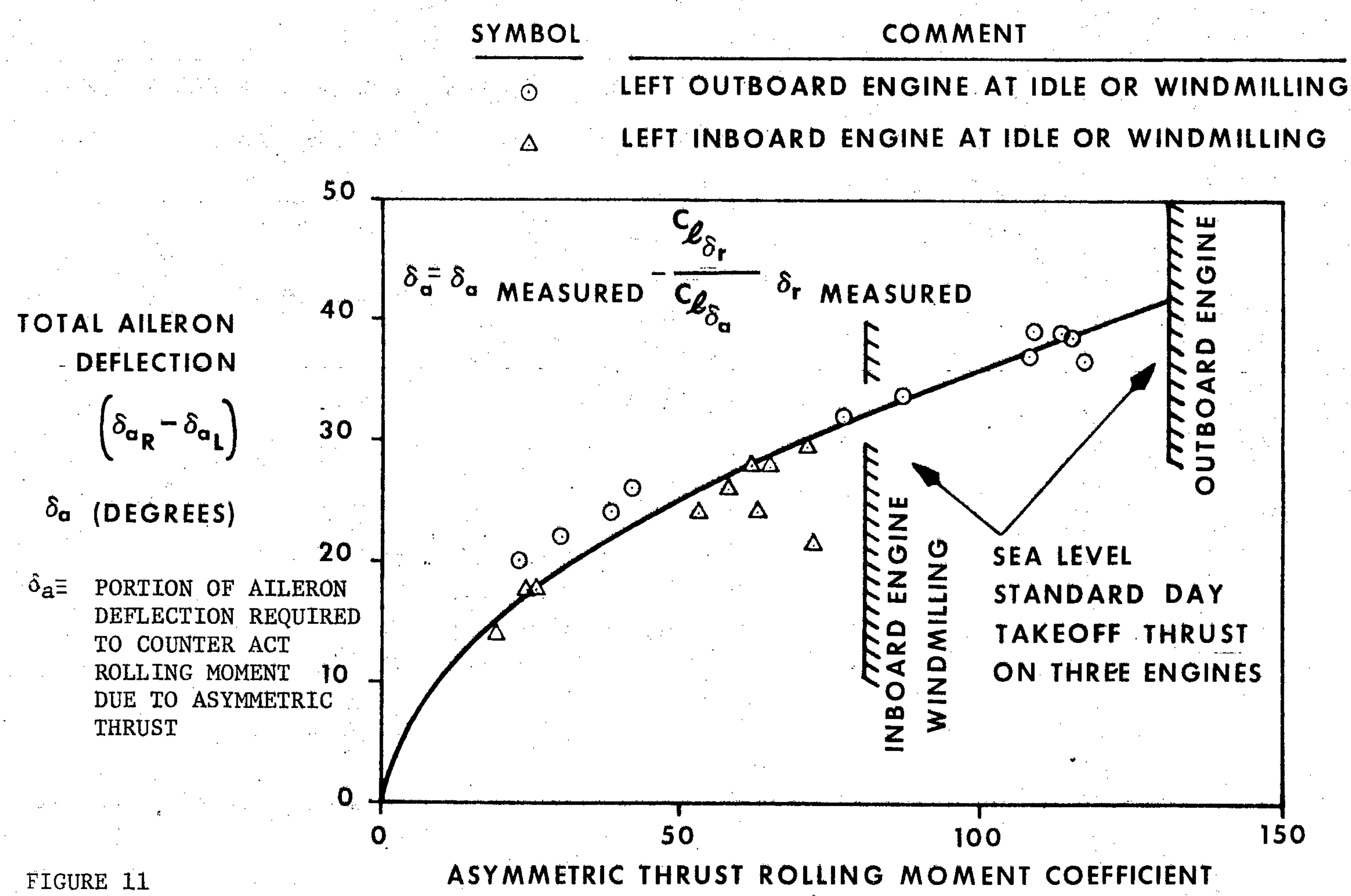


FIGURE 11

Biographical Sketch

Lieutenant David Paul Maunder, was born in Vallejo, California on September 3, 1948. He graduated from Solano Community College, Solano, California in 1971 after five years of part-time study and working as an engineering design draftsman. He graduated from the University of Arizona in 1975, receiving a B.S. degree in aerospace engineering. He was commissioned in the Air Force through Airman Education and Commissioning Program after graduation from OTS. He is nearing completion of an M.S. Degree in aerospace engineering through part-time study at the California State University, Fresno.

In August, 1975 he began his current tour at the Air Force Flight Test Center (AFFTC). His interests have been primarily in the field of parameter identification and automatic controls as applied to aircraft testing. Lieutenant Maunder has been recognized as an innovator of flight test techniques. He is Project Manager of two AFFTC projects; Flying Qualities Criteria, and Aircraft Parameter identification. He is chairman of the AFFTC Parameter Identification Working Group which is active in the enhancement and development of parameter identification flight test techniques.

DEVELOPMENTS IN FLIGHT DYNAMICS TECHNOLOGY FOR NAVY V/STOL AIRCRAFT

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DEVELOPMENTS IN FLIGHT DYNAMICS TECHNOLOGY FOR NAVY V/STOL AIRCRAFT

Abstract

The Naval Air Development Center is undertaking a wide scope of in-house and contractual research efforts directed toward solving the key technical problems associated with estimating and analyzing V/STOL aircraft Flight Dynamics. A major segment of these efforts directly supports the development of a V/STOL Aerodynamics and Stability and Control Manual which will contain up-to-date and evolving methods, data and analysis techniques for predicting V/STOL aircraft stability and control characteristics. The manual format is presented as well as the developmental work flow and details of the supporting study programs. Ongoing V/STOL dynamic response and flying qualities analyses are also briefly described.

Introduction

The Navy is presently committed to developing V/STOL aircraft concepts which will supplement, and eventually supplant, portions of its operational air arm. Proposed missions for these aircraft include launch and recovery operations from small seaborne platforms as well as austere forward sites (Figure 1). The requirement for operations from a variety of air capable ships in day/night and all-weather conditions adds new and unique dynamic problems to those which have previously been identified by land-based V/STOL flight test and research over the past twenty years.

In June 1975, the U. S. Navy V/STOL Technology Assessment (reference (a)) identified V/STOL technology needs and attempted to prioritize them and coordinate efforts to satisfy them. Figures 2a and b summarize the technology need prioritizations in Aerodynamics and Stability and Control resulting from this assessment. It is readily apparent that the highest priority need in both areas is concerned with estimation of aircraft force and moment characteristics. The preliminary designer is unable to satisfactorily estimate V/STOL aircraft static and dynamic characteristics early in the design process. Reference (b) has re-established and further defined this need. The problem is now complicated by the need to estimate these characteristics for aircraft in the presence of a turbulent ship airwake and small moving platform (deck).

The Naval Air Development Center is engaged in a number of in-house and contractual research studies which are aimed at satisfying V/STOL Flight Dynamics technology needs. A major segment of these studies is in direct support of the development of a V/STOL Aerodynamics and Stability and Control Manual. The manual, when completed, will contain up-to-date and evolving methods, data and analysis techniques which will ultimately provide the V/STOL aircraft designer with the capability to rapidly, and with reasonable accuracy, estimate stability and control characteristics and dynamic response parameters of a particular design. These estimates would then form the basis for analyzing flying qualities in the V/STOL flight mode. The remainder of the paper describes the manual format, outlines the developmental work flow and presents details of several of the key supporting study programs. Specific programs include: (1) prediction of induced aerodynamic characteristics during hover in ground effect, (2) measurement of induced forces and moments in a moving deck environment, (3) investigations of jet/free stream flow fields in the transition regime and (4) development of a computerized method to predict aerodynamics in transition/STO flight. A synopsis of major areas of study in V/STOL dynamic response and flying qualities is also presented.

These include: (1) continuing flying qualities research utilizing the X-22A V/STOL aircraft, (2) investigation of the sensitivity of V/STOL designs to flying qualities criteria, (3) investigations of the turbulent flow field in the vicinity of a small ship deck and (4) investigations of launch and recovery dynamics during small ship operations.

V/STOL Stability and Control Manual

The goal of the V/STOL Stability and Control Manual development is indicated in Figure 3. Given preliminary design data on a particular V/STOL concept, it will be possible to rapidly estimate the aerodynamics and stability and control characteristics of the design, from which aircraft dynamic response parameters and flying qualities characteristics can be determined. Principal applications of the manual methods are also listed in Figure 3. These are: (1) rapid assessment of V/STOL configuration and propulsion system alternatives, and trade-offs, at the conceptual design stage; (2) reliable prediction of powered-mode aerodynamics of V/STOL at the preliminary design stage and (3) prediction of stability and control characteristics and trim conditions throughout the V/STOL flight mode.

The V/STOL Manual outline together with an abstract of planned methodologies to be incorporated in it have been reviewed widely within industry and the Government, and the significant suggestions for revision have been included. A broad outline of the manual is shown in Figure 4. As shown, the manual sections are categorized according to V/STOL type consisting of lift/lift-cruise, augmented jet, tilt propellor, tilt-rotor and deflected slipstream. As described in the manual, the designation lift/lift-cruise, section 2.0, encompasses any concept employing combinations of direct lift and/or vectored, deflected or rotated lift/cruise propulsion units. The augmented jet class of V/STOL's, section 3.0, encompasses propulsion systems which employ ejectors for thrust augmentation in the V/STOL flight mode. The format for the tilt-propeller class, section 4.0, is essentially as developed in the present DATCOM. The tilt-rotor class having strong similarities to helicopters in its aerodynamic characteristics is provided a separate section, 5.0. The deflected slipstream class, section 6.0, is intended to be inclusive of emerging or future concepts which deflect jet exhausts over major lifting surfaces. Section 7.0 contains stability and control analysis techniques generally common to all types. Each section contains its own reference list.

Due to wide differences in the aero/propulsive components and methods of predicting aero/propulsive effects of each type, a common outline for each section is not employed. In addition, synopses of current computerized aerodynamic prediction methods employing aircraft and jet modeling routines are included to acquaint the user with these available methods and their capabilities and limitations.

The basic aerodynamic prediction approach, using the manual methods, consists of a component buildup of airframe components including interference effects, propulsion induced aerodynamics whenever separable from the aero/propulsive effects, and effectiveness of aerodynamic and propulsive controls. These methods are to be applied largely in a preliminary design environment but may also be useable to some extent in the early detailed design stage. Within each flight regime, aero/propulsive characteristics can be estimated from which resulting trim characteristics, linear and non-linear stability characteristics at fixed operating points, transition flight trajectories and inputs for piloted flight simulations may be determined. The analysis techniques for stability, control and flying qualities are contained in section 7.0. It is intended that the estimates obtained be used for evaluating aircraft designs, for determining stability and control and flying qualities to a reasonable level of accuracy and for highlighting problem areas.

The planned developmental work flow for the manual is illustrated in Figure 5. In addition to the review and critiques already mentioned, a data bank of methods and test data serves as a starting point for the manual development. A feedback from this work flow will be a continuing awareness of the limitations and gaps in the technology for which research efforts must be expended to fill the gaps. The intent is to incorporate the current most reliable semi-empirical methods for estimating aero/propulsive stability and control characteristics of a wide variety of V/STOL classes for the major regions of V/STOL flight. These are (1) hover in and out of ground effect, (2) transition, and (3) STOL. The methods will consist of empirical charts and formulations for easy application and will be maintained current by periodic revision.

The manual is to be used in conjunction with the USAF DATCOM. Applicable sections of DATCOM are identified in the manual to be used to determine aerodynamics of certain basic unpowered components (e.g., fuselage) for completeness in a total vehicle component buildup.

Hover Aerodynamics

Typical of the empirical methods to be incorporated in the manual is one empirical formulation developed by McDonnell Aircraft Company for the jet induced lift loss of V/STOL aircraft hovering out of ground effect--sometimes referred to as a base loss--shown in Figure 6. Test data from several sources utilizing single and multiple jets were found to correlate with the expression:

$$\frac{\Delta L/T}{\sqrt{S/A_j}} = -.0002528[NPR^{-.64}(\Sigma P/D_e)]^{1.581}$$

as shown in the figure. This is a somewhat simpler method to apply than the earlier well known method of Margason (reference (c)) which requires knowledge of the jet dynamic pressure decay rate. Implicit in the expression is an accounting for the increased entrainment of multiple jets by the terms involving the summation of jet perimeters, $\Sigma P/D_e$, which was also a finding in the Margason tests. Generally the magnitude of out-of-ground-effect hover lift loss for a typical VTOL is only of the order of 2 to 3%. However it can be a significant factor in engine sizing for some VTOL types.

Generally the larger area of concern with lift loss occurs when hovering in-ground-effect because of the increased entrainment of the ground flow that is encountered. The well known method of Wyatt (reference (d)) for entrainment induced lift loss of single jets has been modified and extended to be applicable to the multiple jet case. Figure 7 shows test data from Margason (reference (c)), Vogler (reference (e)) and Louisse and Marshall (reference (f)) for single and multiple jets having various ratios of planform to jet area (S_p/A_j), correlated with the Wyatt curve for single jets. In the Wyatt correlation, h is the height of the planform above ground and $\overline{D}$ is the equivalent diameter of the planform surface. Good correlation is shown, although a slightly lower slope than the Wyatt curve at low $h/\overline{D}$ is indicated by the data. The implication of these results is that, at least for closely spaced multi-jet configurations (which was the case in the tests listed on Figure 7), a good estimate of entrainment lift loss can be obtained from the modified Wyatt curve by a simple superposition of the lift loss caused by each jet in the configuration. One application of this modified Wyatt technique using a superposition approach was made for the AV-8A Harrier configuration and compared with model test data in Figure 8. Good lift loss correlation is shown except where the fountain effect appears in the test data for this four jet configuration below $h/D_e = 5$.

The fountain upload effect is another important area to be understood and predicted so that the entire ground effect can be treated in a building block approach. An empirical prediction method for the fountain induced upload which will account for the influence of undersurface shapes and appendages is being sought for inclusion into the manual.

In addition to methods based on pure empiricism, computerized potential flow techniques using empirical jet models are also being developed. One such recently developed computerized technique by McDonnell Aircraft Company sponsored by the Naval Air Development Center is summarized in Figure 9. In this program the influence of the different flow regions i.e., free jets, wall jets and fountain flows on the aircraft surfaces are determined. The aircraft surfaces are divided into panels on which the Douglas Neuman potential flow solution is satisfied. Added to this are the paneled boundaries of the free and wall jet with predetermined amounts of inflow velocities due to entrainment. The program routine also computes the ground flow stagnation lines for arbitrary numbers and location of jets as well as the distribution of upflow momentum in the fountain. The lower right hand figure shows the solution for the total momentum flux available in the fountain as a fraction of the issuing jets thrust. In general only a small portion of this momentum is converted into positive uploads on aircraft surfaces due to interactions between the free jets and the fountain and due to an additional entrainment caused by the fountain itself. Further detailed test investigations of the fountain properties are needed as a basis for improving the models that are incorporated in present computer programs.

Another area being actively investigated in V/STOL hovering flight is the effect of a moving deck on the jet induced forces and moments. A test and analysis program to investigate these time dependent dynamic effects is nearing completion. Figure 10 illustrates the problem being investigated and the program objectives. The heaving, pitching, and rolling motions of a seaborne platform were approximated while the induced forces and moments acting on both fully-contoured and planform models of V/STOL aircraft were measured. The objectives of the program were to assess the jet-induced aerodynamics to determine the frequency content of the aerodynamic data and the phase relationship between the induced forces and moments and the deck motion, and to empirically formulate the time dependent induced forces and moments. The model configurations employed and the general test ranges are given in Figure 11. Two types of models were used in the test investigation: a subsonic configuration tested in full contour and as a flat plate and a supersonic model tested as flat plate.

In the test program, the effects of the following parameters on the induced aerodynamics were investigated: (1) model contouring (3-D vs. planform), (2) nozzle vectoring vanes, (3) nozzle arrangement, (4) nozzle operating conditions, (5) wing height, (6) lift improvement devices, (7) deck position and motion, and (8) deck size. The majority of testing was accomplished with dynamic deck motion but data were obtained at static hover conditions, as well, for comparison purposes.

A preliminary comparison between the induced lift with heaving deck motion and the induced lift variation with deck height as derived from static hover data is shown in Figure 12 for a case where the deck neutral point is near the point of peak fountain lift. The variation derived from static data was obtained by entering the static hover induced lift curve at appropriate heights corresponding to given instants in time. For this case, the induced lift variations are somewhat similar in form but fairly significant differences can be seen in the amplitude of the induced lift fluctuation. A similar comparison of the rolling moment variation with deck roll is shown in Figure 13. As with the induced lift, the measured rolling moment variation and the variation derived from static data are similar in form but the amplitudes differ. The amplitude differences in the induced force and moment data were found to be frequency dependent from tests where the deck motion frequency was varied from 1 to 3 Hz. Comparisons of the type presented above are being used to assess the applicability of static hover data to the prediction of the time-dependent induced aerodynamics. In addition, empirical methods of predicting the dynamic induced force and moment variations are being formulated. Several combined motions, e.g., pitch/roll and heave/roll, and simulated take-off and landing sequences were also included in the dynamic tests to determine the resultant induced aerodynamics under these complex situations.

Sufficient dynamic force balance data were obtained to allow statistical analyses of the induced force and moment data. Power spectral densities are being used to evaluate the frequency content of the data while cross-power spectral densities are being used to correlate the output and driving functions and to determine the phase relationship between the two. Empirical formulations of time dependent forces and moments is intended for inclusion in the V/STOL Manual.

Transition Aerodynamics

Areas of transition aerodynamics being studied include a joint test program with NASA Langley to obtain detailed flow surveys of rectangular jets in a crossflow. The test setup is illustrated in Figure 14. Items being measured are the jet centerline path, the path of the contra-rotating vortices that develop along the jet, the strength of these vortices and detailed pressure surveys on the plate containing the issuing jet. The rectangular jet was tested in the orientation shown, and in a streamwise orientation with the y axis rotated 90°. Jet inclinations (measured from horizontal) varied from 45° to 105° (15° forward of vertical). This test is a continuation of earlier tests to obtain this same type of data for circular jets. One purpose is to provide an empirical basis for developing math models to represent the

flow properties (the contra-rotating-vortex strength and entrainment rates) as an input to the computerized potential flow models that are being developed for predicting aerodynamics of V/STOL aircraft in transition. The measured surface pressure distributions on the issuing plate provide not only a basis for validation of the computerized aerodynamic techniques, but can also be used as the basis for developing an empirical prediction technique for jet induced forces and moments in transition flight. Figure 15 shows typical contours of surface pressure coefficients measured on a flat plate due to a circular jet issuing into a crossflow ($\delta_j = 90°$), overlayed on a representative aircraft configuration. The measured pressure coefficients $C_p = (P_s - P_\infty)/q_\infty$ have a functional variation with the normalized x/D, y/D dimensions as well as with the velocity ratio, V_∞/V_j, and the inclination of the jet to the freestream flow, δ_j. An in-house effort is attempting to develop polynominal expressions for these functional relationships such that the induced lift and moment can be obtained by simple integration of the pressure coefficient relations over any arbitrary planform of interest. Induced lift normalized to jet thrust will then be computed by:

$$\frac{\Delta L}{T} = \frac{1}{\pi}\frac{q_\infty}{q_j}\int_S C_p \; d(x/D)\,d(y/D)$$

Complete configuration effects can be obtained by a buildup of single jet effects.

For the 90° injection case, a reasonably good curve fit to the available data has been found by obtaining an expression for Cp_{max} as a function of velocity ratio and spanwise dimension from the jet orifice, $Cp_{max} = f(V_\infty/V_j,\ y/D)$. The C_P values are then developed as an exponential decay from Cp_{max} in the x/D coordinate.

Computerized Methods

In the development of new aircraft of unconventional type, model testing has played a major role in all aspects of aerodynamic analysis. It is apparent, however, that computerized techniques of aerodynamic analysis and prediction may assume a wider usage in the future. The development of V/STOL aircraft seems to be an example for which

computerized aerodynamic analysis techniques may play a more decisive role. One problem with computerized techniques in the past has been the cumbersome and time consuming data input and the high computing times, especially when examining characteristics of a variety of design concepts. Considerable progress has and is being made to improve input and computational efficiency. The Naval Air Development Center is supporting efforts to improve several areas of a computer program originally developed by Vought aircraft for transition aerodynamics. The input and computational routines of the existing program are shown in the flow diagram of Figure 16. The geometry input involves paneling of aircraft surfaces as required by the Hess aerodynamics potential flow routine. The jet effects program module is as developed by Wooler-Ziegler (reference (g)), which models the jet as a series of sinks and doublets. Nacelle or engine inlet effects are treated by the Stockman (NASA Lewis) engine inlet routine with modifications incorporated for boundary layer growth. Modifications and extensions of the program will consist of (1) incorporation of a geometry data input routine, (2) an extension to incorporate the Fearn and Weston vortex model for the jet, as well as the models being developed from the rectangular jet test program previously mentioned, (3) modifications to the entrainment and vortex models close to the jet exhausts to improve the prediction of pressures in the jet wake region where viscous effects are clearly dominant, and (4) incorporation and modification of routines to predict the effects of ground proximity at STOL speed and thrust conversion conditions.

The basic vortex and entrainment model developed by Fearn and Weston is illustrated in Figure 17. It places filament vortices along the pre-determined vortex path and sink distributions on the pre-determined centerline. As mentioned, modifications to the local entrainment rate and vortex strengths are made to improve predictions in the wake region. The program, when completed, will be exercised against a variety of configuration models for validation. It will be synopsized in the V/STOL Stability and Control Manual and made available to industry.

V/STOL Flying Qualities Developments

In addition to being able to estimate the aerodynamic stability and control characteristics of a given V/STOL configuration, the aircraft designer must also be able to distinguish between "good" and "bad" characteristics in the expected flight environment. To do this successfully he must have adequate stability and flying qualities guidelines and criteria as well as representative models of air turbulence and mean wind conditions which might be encountered. Likewise, in order to investigate design tradeoffs, he needs information regarding the sensitivity of the aircraft design to changes in the specified criteria.

Finally, in the case of small shipboard operations, he must be able to consider the dynamic interaction between the ship and the aircraft. Studies addressing these requirements are briefly discussed in the following.

X-22A In-Flight Research

The X-22A Variable Stability V/STOL Research Aircraft (Figure 18) is being used to provide in-flight data for development and validation of V/STOL flying qualities criteria. The aircraft is equipped with a hybrid (analog/digital), full authority variable stability/variable control system coupled with a programmable head-up-display unit. The overall system provides an ideal base for varying and documenting the effects of a wide range of dynamic parameters inflight (hover through transition). Calspan Corporation maintains and operates the aircraft at the Buffalo airport under Navy contract. Experiment planning and post-flight analyses are performed jointly by Navy and Calspan personnel.

Task V, the current flight research program, is addressing V/STOL flying qualities criteria for both VFR and IFR conditons in the hover and transition flight regimes. Task V is divided into three sub-tasks. The Sub-task (a) experiment, which is now in the detailed planning stage, will investigate response and control power requirements for TRCS (Translational Rate Control Systems) in the hover/low-speed regime in a VFR environment. The flight experiment will be conducted with and without simulated turbulence using a series of flight tasks which approximate the level of difficulty of terminal operations relative to a DD963 class ship. Specifically the experiment is being designed to quantify response characteristics for systems using attitude control and direct force control, both with and without control power limiting. Experiment design is relying heavily on existing simulator and flight data (e.g., Figure 19) for relative parameter significance and range definitions in the hope that in-flight results can be used to validate/"fine-tune" existing ground-based results.

Sub-tasks (b) and (c) are currently planned to address IFR hover requirements and transition dynamic requirements, respectively. The entire Task V experiment is scheduled to run through September of 1980.

V/STOL Design Sensitivity to Flying Qualities Criteria

In V/STOL aircraft design (as in any aircraft design) the final configuration is optimized to efficiently and effectively meet mission requirements through a series of parametric trade studies. Flying qualities criteria may have a much greater impact on V/STOL concept

sizing than normally experienced for CTOL design due to additional requirements which are imposed on the primary propulsion and control systems. If these additional requirements are more demanding than those imposed by the primary performance constraints the design will be penalized accordingly. The magnitude of the penalty must be known if a reasonable compromise between flying qualities requirements and performance requirements is to be achieved.

Vought Corporation, under the cognizance of the Naval Air Development Center, is beginning a one year study which will attempt to quantify these sensitivities. Baseline subsonic and supersonic V/STOL configuration will first be "point-designed" to meet mission oriented performance requirements. Sensitivities will be determined for preselected flying qualities criteria by comparing the baseline designs with designs sized to meet critiera which vary in both magnitude and form. Performance, weight, cost and developmental risk factors will be considered. Figure 20 depicts the planned study flow as well as general sketches of the proposed baseline designs.

Small Ship Airwake Modelling

The steady and random components of the airwake downwind of the landing platform of a small V/STOL capable ship (e.g., a DD 963 destroyer) may have considerable impact on the controllability of a V/STOL approaching and landing or taking-off. A realistic quantified model of the airwake is required in conjunction with the aerodynamic model of the aircraft to adequately design a vehicle which stands a good chance of success in this environment.

In order to quantify the airwake, several ship configurations have been, or are scheduled to be, tested in the V/STOL Wind Tunnel at Boeing Vertol Company. The test set-up is shown in Figure 21. The ship model is positioned at various combinations of yaw and roll angle relative to the wind direction and the X, Y, and Z components of the airwake are measured using a grid of extremely sensitive "split-film" hot wire anemometer probes. Probe locations are determined based on smoke flow visualization techniques which define the principal areas of turbulence in the airwake. The resulting data are analyzed to determine the mean and random components of the velocity field, properly scaled based on Strouhal Number ($\omega \ell/V$) and resolved into components. Figure 22 is representative of the data thus obtained for an FF 1052 class frigate (reference (h)). Preparations are currently underway for a similar test on a DD 963 class destroyer in October/November 1978.

Once the data have been measured in the wind tunnel and analyzed to determine the steady and non-steady characteristics, they must be reduced to a tractable form for analysis and simulation. Initial analysis of the data indicated that both the mean velocity ($\bar{V}$) and standard deviation about the mean (σ) scale reasonably well with the magnitude of the wind over the deck (V_{WOD}). Both velocities are, however, strongly dependent on position relative to the ship and wind direction (as would be expected). Fast Fourier Transform analysis of the time dependent data was used to determine its frequency domain characteristics. The results suggested that the random airwake components could be simulated by passing white noise through a first order filter with a gain proportional to the measured standard deviation and a break frequency proportional to the magnitude of the wind over deck (Figure 23). Reference (i) details the results of the modelling effort.

Launch and Recovery Dynamic Analyses

In order to predict the performance and dynamic characteristics of a V/STOL aircraft operating in the vicinity of a small V/STOL-capable ship, dynamic models of all the active and passive elements must be developed and their interactions must be determined. The elements to be considered include: (1) aircraft aerodynamics, (2) engine and reaction control dynamics, (3) stability augmentation and control systems, (4) ship dynamics, (5) atmospheric and ship induced turbulence, (6) pilot dynamics, (7) visual display and landing aid dynamics and (8) LSO (Landing Signal Officer) interface. Figure 24 depicts the interaction of the various elements. Previously described efforts address developments in several of these elements. The following briefly describes two methods by which these element models are combined and utilized for dynamic analysis.

A time domain (time history) analysis program has been developed which addresses the dynamic characteristics and interactions of the above elements (reference (j)). The program, within the limitations of the assumed pilot model, may be used to investigate vehicle sensitivities to various element characteristics and noise. All results are computed as time histories/ trajectories.

A second method has been developed by Vought Corporation under contract to the Naval Air Development Center. This method utilizes a Covariance Propogation Technique to analyze the statistical implications of various disturbances to the system elements. Using this approach, credible statistical results may be obtained without having to compute a large number of time histories. Figure 25 graphically describes the differences between the two methods.

In the case of both methods the primary framework of the analysis has been developed. Many of the individual component models, however, require further development and refinement. Work is both ongoing and planned to improve these models.

Concluding Remarks

In summary, the Naval Air Development Center is developing V/STOL aerodynamics, stability and control prediction capability through several in-house and contracted studies. The effort will ultimately result in a V/STOL Aerodynamics and Stability and Control Manual providing the aerospace community with rapid and reliable prediction methods. The initial release of the Manual is scheduled for 1979 with future revisions and updates occurring as necessary.

Acknowledgement

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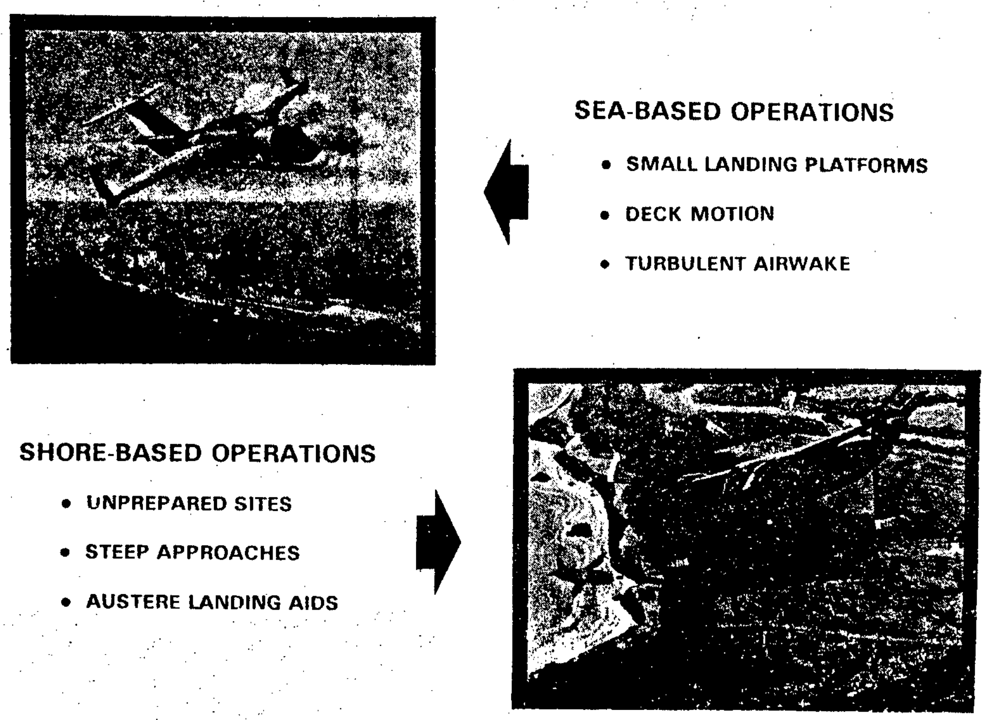


FIGURE 1. PROPOSED V/STOL OPERATING ENVIRONMENTS

TECHNOLOGY ITEMS / GENERIC VEHICLES	WIND TUNNEL/ FLIGHT DATA COLLECTION	PREDICTION OF FORCES & MOMENTS COMPTS. PARTIALLY IMMERSED	SCALING FACTORS & TUNNEL COR-RECTIONS	CORRELATIVE PER-FORMANCE PARAMETERS	CENTER OF PRESSURE PREDICTION AUGMENT-ERS/FANS	OPTIMAL HIGH-SPEED DESIGN MINIMIZING LOW-SPEED PENALTIES
JET-LIFT	1 / 1	2 / 6	3 / 3	1 / 1	N/A / N/A	4 / 1
TILT WING	2 / 2	1 / 1	4 / 5	1 / 4	N/A / N/A	3 / 5
TILT PROP/ROTOR	3 / 5	1 / 2	2 / 6	1 / 5	N/A / N/A	4 / 6
DEFLECTED SLIPTREAM	3 / 7	1 / 7	2 / 7	1 / 7	4 / N/A	5 / 7
DUCTED PROP	1 / 4	2 / 3	3 / 4	1 / 6	5 / N/A	4 / 4
AUGMENTER	4 / 6	1 / 5	3 / 1	1 / 2	2 / 1	5 / 2
FAN-IN-WING/ BODY	2 / 3	1 / 4	3 / 2	1 / 3	4 / 2	5 / 3

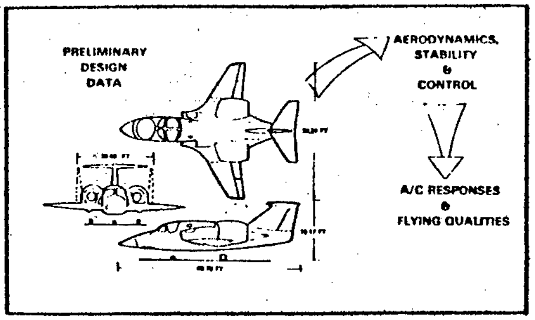
A/B: A = TECHNOLOGY PRIORITY WITH VEHICLE TYPE (READ ACROSS)
B = VEHICLE PRIORITY WITHIN TECHNOLOGY (READ DOWN)

FIGURE 2a. V/STOL AERODYNAMICS: TECHNOLOGY NEEDS/PRIORITIES

TECHNOLOGY ITEMS / GENERIC VEHICLES	ESTIMATION TOTAL AIRCRAFT DERIVATIVES	E.O.M. PARAMETER SENSITIVITY	INTEGRATED VSTOL DYNAMIC ELEMENTS MODEL	VSTOL DESIGN SENSITIVITY TO S&C PARAMETERS	FLYING QUALITIES SPECIFICATION REVISION	VSTOL PILOTED SIMULATION	FLIGHT-DECK TURBULENCE & MOTION	FLYING QUALITIES RESEARCH IN TRANSITION	SEA-CAPABLE VARIABLE STABILITY RESEARCH VEHICLE	VSTOL PARAMETER IDENTIFICATION	CONTROLS/PROPULSION INTEGRATION	DESIGN CRITERIA AXES COUPLING	VSTOL PILOTED-INDUCED OSCILLATIONS	VSTOL FLEXIBLE DYNAMICS: AERO-ELASTIC INSTABILITIES	VSTOL BUFFET & STALL DURING TRANSITION
JET LIFT	1 / 1	2 / 1	7 / 1	3 / 1	8 / 1	4 / 1	N/A / N/A	5 / 3	9 / 3	6 / 1	10 / 3	11 / 3	12 / 3	13 / 6	14 / 6
TILT WING	1 / 4	2 / 4	7 / 4	3 / 4	6 / 4	5 / 4	N/A / N/A	4 / 5	9 / 5	8 / 2	10 / 5	11 / 4	12 / 4	13 / 2	14 / 1
TILT PROP/ ROTOR	1 / 5	2 / 5	7 / 5	3 / 5	6 / 5	5 / 5	N/A / N/A	8 / 6	9 / 6	10 / 5	11 / 6	12 / 5	13 / 5	4 / 1	14 / 5
DEFLECTED SLIPSTEAM	1 / 7	2 / 7	3 / 7	4 / 7	5 / 7	6 / 7	N/A / N/A	7 / 7	8 / 7	9 / 7	10 / 7	11 / 7	12 / 7	13 / 7	14 / 7
DUCTED PROP	1 / 6	2 / 6	7 / 6	3 / 6	6 / 6	5 / 6	N/A / N/A	8 / 4	9 / 4	10 / 6	11 / 4	12 / 6	13 / 6	4 / 3	14 / 2
AUGMENTER	1 / 3	2 / 3	5 / 2	3 / 2	6 / 2	4 / 2	N/A / N/A	7 / 1	8 / 1	9 / 3	10 / 1	11 / 1	12 / 2	13 / 4	14 / 3
FAN-IN-WING/ BODY	1 / 2	2 / 2	5 / 3	3 / 3	6 / 3	4 / 3	N/A / N/A	7 / 2	8 / 2	9 / 4	10 / 2	11 / 2	12 / 1	13 / 5	14 / 4

A/B: A = TECHNOLOGY PRIORITY WITHIN VEHICLE TYPE (READ ACROSS)
B = VEHICLE PRIORITY WITHIN TECHNOLOGY (READ DOWN)

FIGURE 2b. V/STOL STABILITY AND CONTROL: TECHNOLOGY NEEDS/PRIORITIES



APPLICATION

- RAPID ASSESSMENT OF V/STOL CONFIGURATION AND PROPULSION SYSTEM ALTERNATIVES, AND TRADE-OFFS, AT THE PRELIMINARY DESIGN STAGE
- RELIABLE PREDICTION OF POWERED MODE AERODYNAMICS OF V/STOL AIRCRAFT AT THE PRELIMINARY DESIGN STAGE
- PREDICTION OF STABILITY AND CONTROL CHARACTERISTICS AND TRIM CONDITIONS THROUGHOUT THE V/STOL MODE

FIGURE 3. V/STOL MANUAL APPLICATIONS

1.0 FORMAT DESCRIPTION

2.0 LIFT/LIFT-CRUISE FAN/JET V/STOL'S

- 2.1. BASIC (UNPOWERED) AERODYNAMIC CHARACTERISTICS
  - 2.1.1. FUSELAGE
  - 2.1.2. LIFTING SURFACES
- 2.2. PROPULSION INDUCED AERODYNAMICS
  - 2.2.1. HOVER
    - 2.2.1.1. FORCE & MOMENT VARIATIONS WITH GROUND HEIGHT
    - 2.2.1.2. EFFECTS OF PITCH & ROLL ATTITUDE
    - 2.2.1.3. EFFECT OF APPENDAGES
    - 2.2.1.4. FORCE & MOMENTS DUE TO TRANSLATIONS OR CROSSWINDS (OGE)
    - 2.2.1.5. INLET MOMENTUM EFFECTS
    - 2.2.1.6. MOVING DECK DYNAMIC EFFECTS
    - 2.2.1.7. COMPUTERIZED MODELING TECHNIQUES
  - 2.2.2. TRANSITION
    - 2.2.2.1. LONGITUDINAL
    - 2.2.2.2. LATERAL-DIRECTIONAL
- 2.3. PROPULSION SYSTEM CHARACTERISTICS
- 2.4. CONTROL CHARACTERISTICS
  - 2.4.1. AERODYNAMICS CONTROLS
  - 2.4.2. AERO/PROPULSION CONTROLS

3.0 AUGMENTED JET V/STOL'S

4.0 PROPELLER V/STOL'S

5.0 ROTOR V/STOL'S

6.0 DEFLECTED SLIPSTEAM V/STOL'S

7.0 STABILITY & CONTROL ANALYSIS TECHNIQUES

- 7.1. EQUATIONS OF MOTION
- 7.2. TRIM CALCULATIONS
- 7.3. STABILITY DERIVATIVE ESTIMATION
- 7.4. SMALL PERTURBATION STABILITY ANALYSIS
- 7.5. COMMON CONTROL TRANSFER FUNCTIONS

FIGURE 4. V/STOL MANUAL OUTLINE

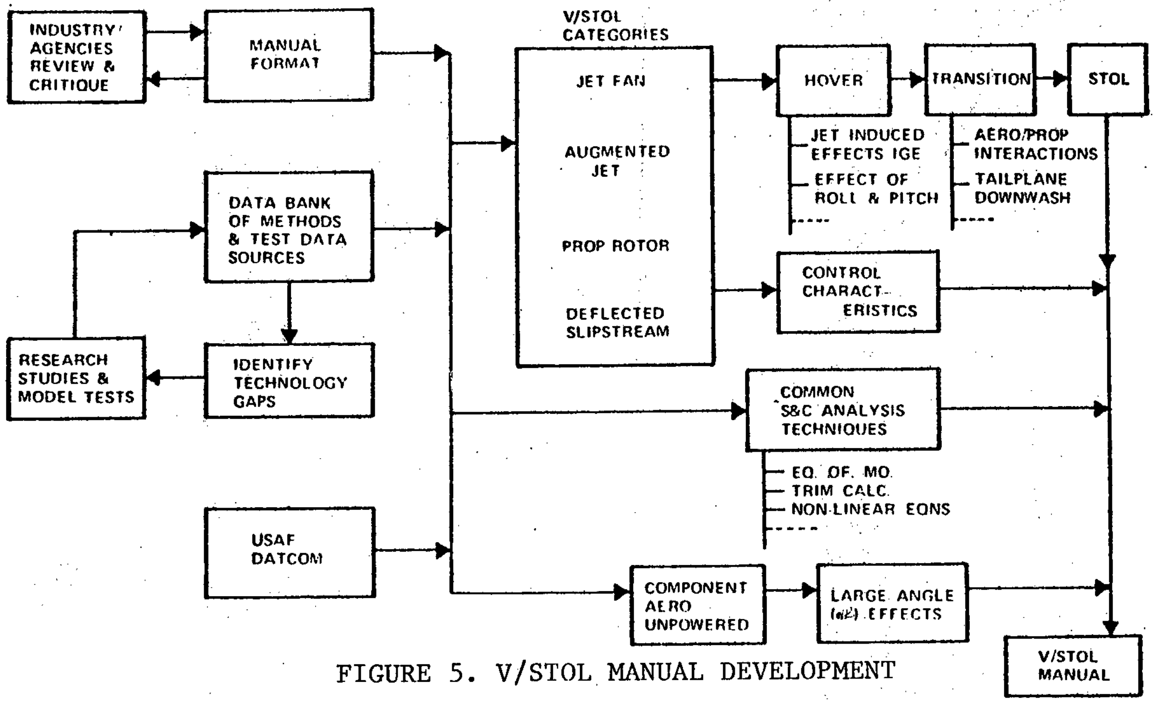


FIGURE 5. V/STOL MANUAL DEVELOPMENT

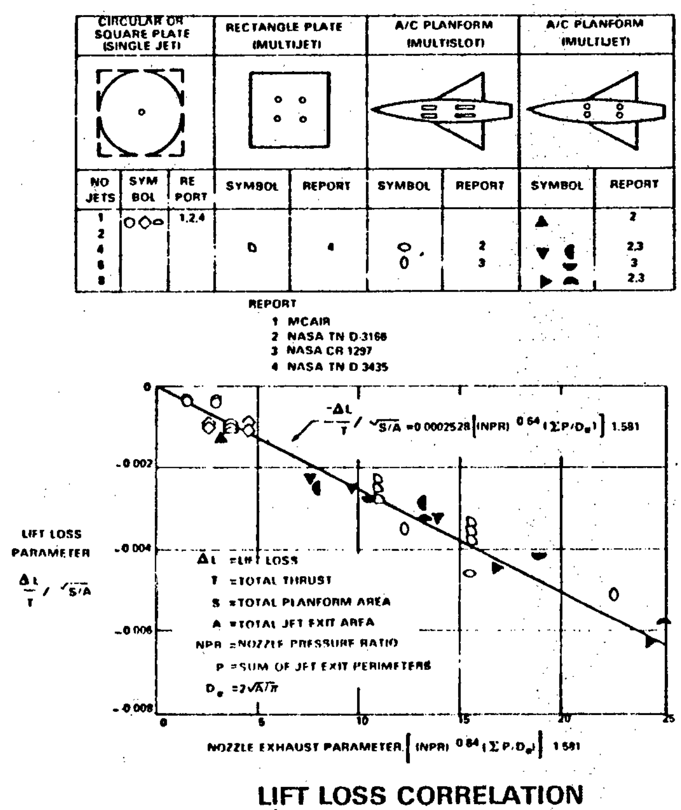


FIGURE 6. LIFT LOSS IN HOVER (OGE)

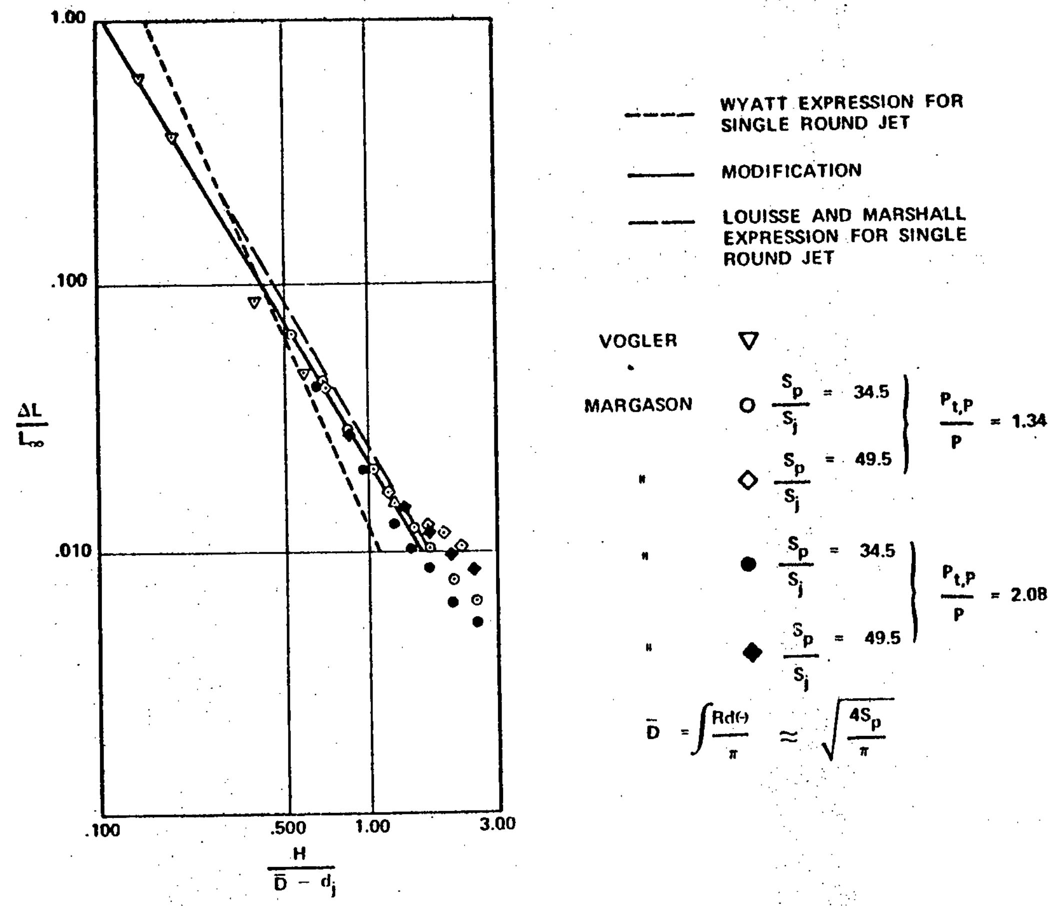


FIGURE 7. LIFT LOSS IN HOVER (IGE)

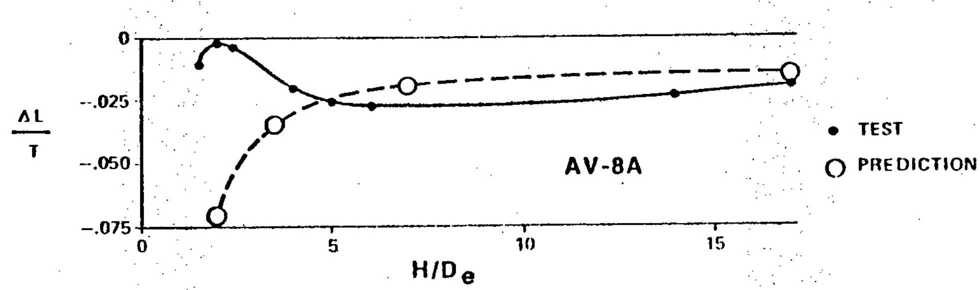


FIGURE 8. INDUCED LIFT LOSS: SUPER-POSITION APPROACH

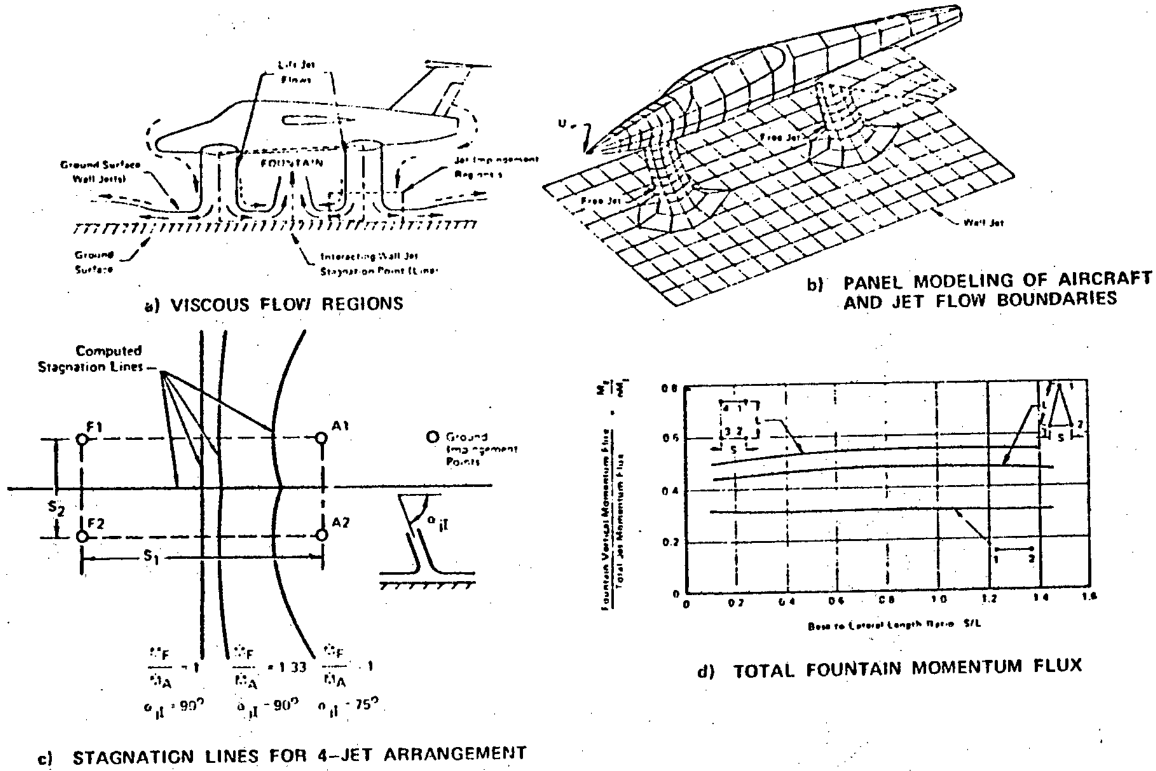
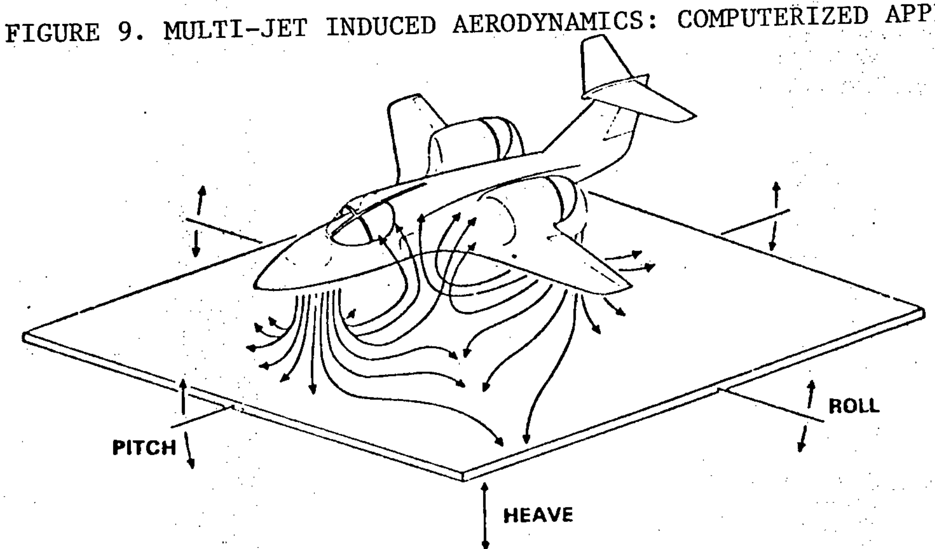


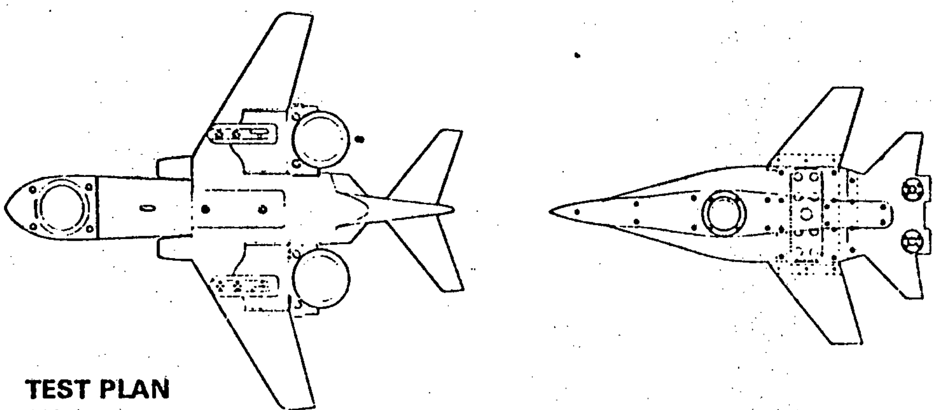
FIGURE 9. MULTI-JET INDUCED AERODYNAMICS: COMPUTERIZED APPROACH FOR HOVER



PROGRAM OBJECTIVES

- **TO ASSESS THE JET-INDUCED AERODYNAMICS OF V/STOL AIRCRAFT HOVERING OVER A MOVING DECK**
- **TO DETERMINE FREQUENCY CONTENT OF THE INDUCED FORCE AND MOMENT DATA AND THE PHASE RELATIONSHIP BETWEEN THE FORCES AND MOMENTS AND THE DECK MOTION**
- **EMPIRICALLY FORMULATE TIME DEPENDENT INDUCED FORCES AND MOMENTS ACTING ON V/STOL A/C IN PROXIMITY OF MOVING DECK**

FIGURE 10. INDUCED AERODYNAMICS IN A MOVING DECK ENVIRONMENT: PROGRAM OBJECTIVES



TEST PLAN SUMMARY:

TEST FULLY CONTOURED AND FLAT PLATE SUBSONIC V/STOL CONFIGURATION AND FLAT PLATE SUPERSONIC V/STOL CONFIGURATION OVER RANGES OF AMPLITUDES AND FREQUENCIES FOR A VARIETY OF DECK MOTIONS

GENERAL TEST RANGES

H/D_E:	0.5 TO 10
$\Delta h/D_E$:	$\pm$ 0.5 TO $\pm$ 1.5
$\Delta\alpha$:	$\pm 2^\circ$ TO $\pm 10^\circ$
$\Delta\gamma$:	$\pm 2^\circ$ TO $\pm 10^\circ$
f:	0 to 3 Hz
NPR:	1.1 TO 2.5

FIGURE 11. INDUCED AERODYNAMICS IN A MOVING DECK ENVIRONMEMT: TEST MODEL DESCRIPTIONS

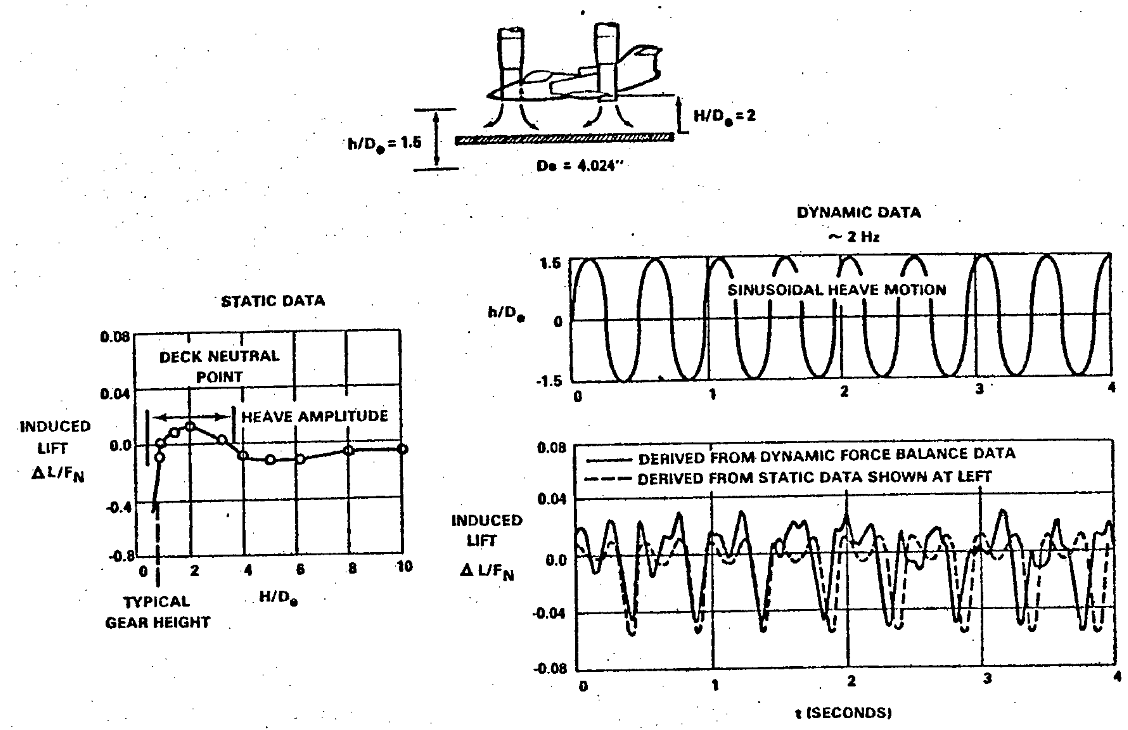


FIGURE 12. DYNAMIC INDUCED LIFT (HEAVE MOTION)

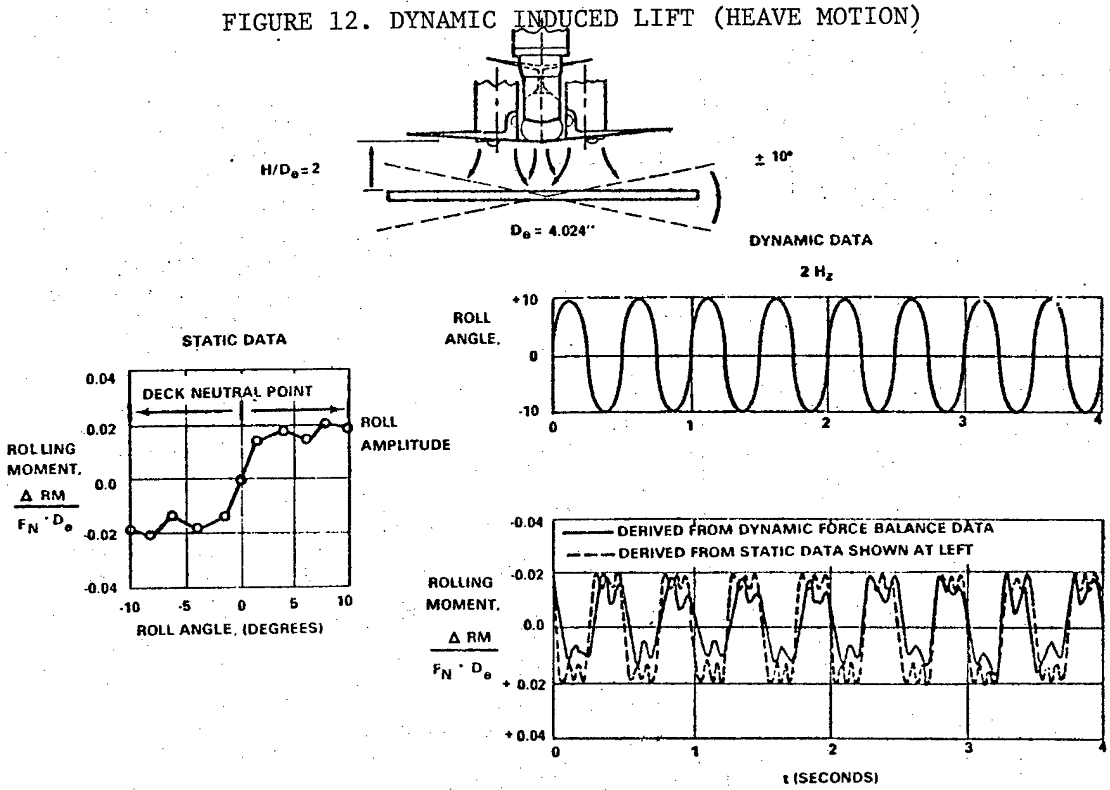


FIGURE 13. DYNAMIC INDUCED LIFT (ROLL MOTION)

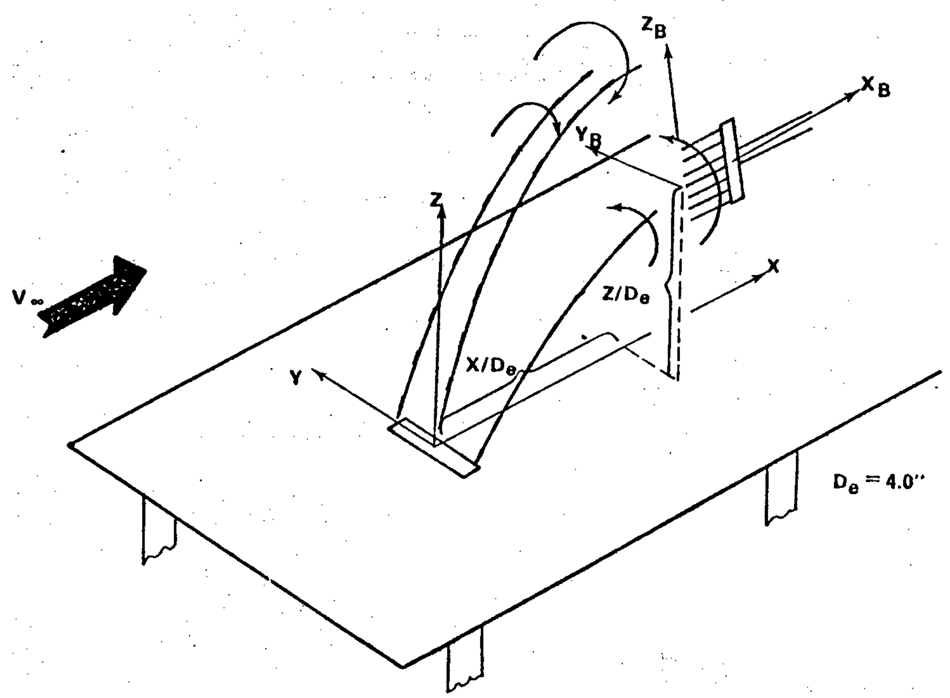


FIGURE 14. RECTANGULAR JET CHARACTERISTICS: TEST SETUP

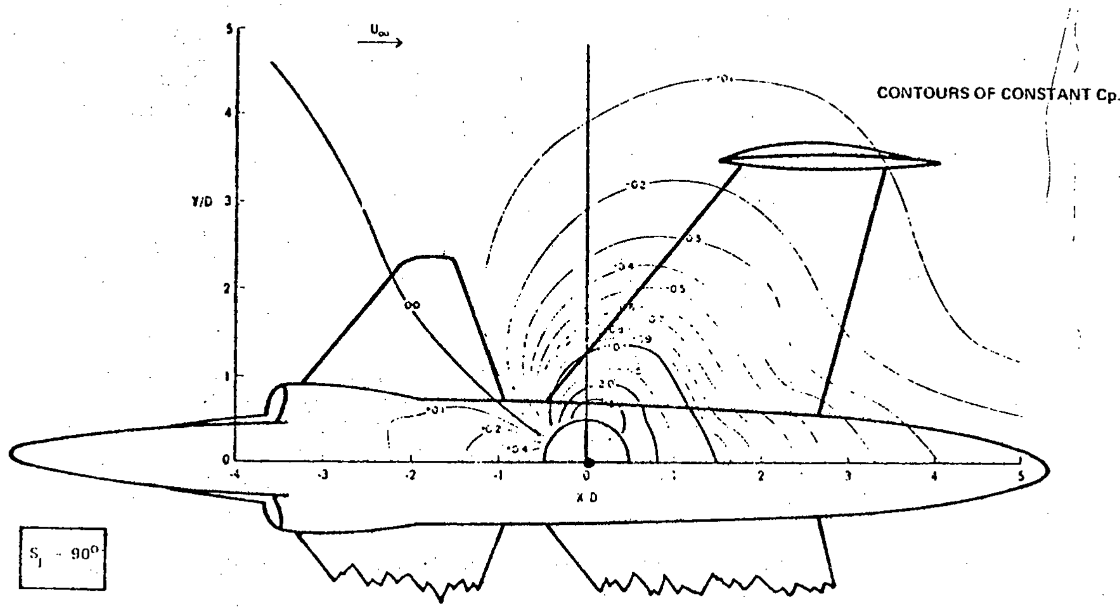


FIGURE 15. JET INDUCED SURFACE PRESSURE CONTOURS

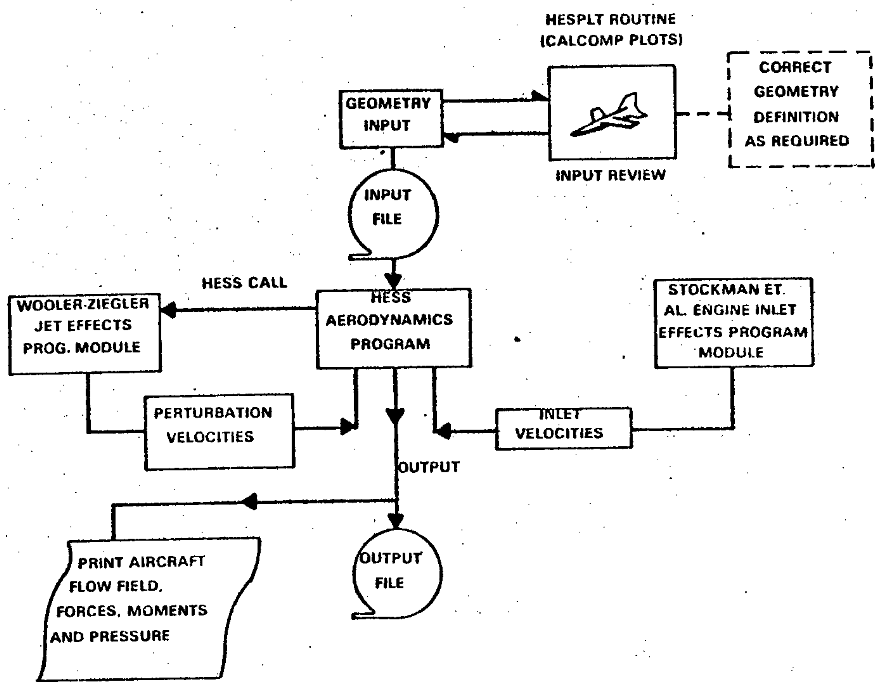
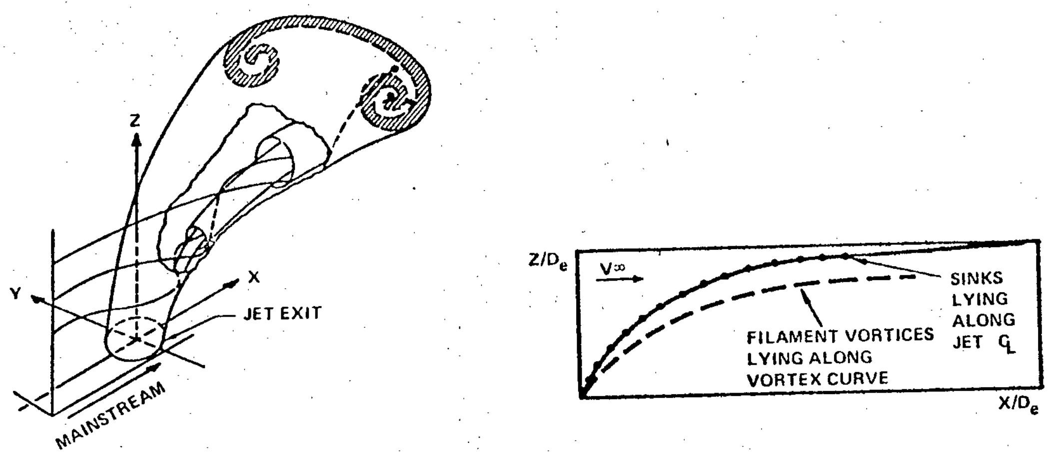


FIGURE 16. MULTI-JET INDUCED AERODYNAIMCS: COMPUTERIZED APPROACH FOR TRANSITION

THE VAPE PROGRAM WILL BE EXTENDED TO INCLUDE CALCULATIONS OF THE TRAJECTORY AND STRENGTH OF THE CONTRA-ROTATING VORTICES FORMED BY A PROPULSIVE JET ISSUING INTO A CROSSFLOW



SCHEMATIC OF FORMATION OF CONTRA-ROTATING VORTICES CREATED BY THE JET ISSUING INTO A CROSSFLOW

FERN AND WESTON'S MODEL OF A CIRCULAR JET IN A SUBSONIC CROSSFLOW

FIGURE 17. FEARN AND WESTON VORTEX AND ENTRAINMENT MODELS

GENERAL SPECIFICATIONS

DIMENSIONS

Length	39.57 ft	
Height	20.69 ft	
Tread	8.0 ft	
Wing	Front	Aft
Area	139 sq ft	286 sq ft
Span	22.97 ft	39.24 ft
Aspect Ratio	3.86	5.38

ENGINE RATINGS

SHP	SLS	Thrust	rpm	Min.
1250	Mil.	154	19,500	30
1050	Nor.	132	19,500	Cont.

POWER PLANT

No. & Model	(4) YT58-GE-8D
Mfr.	General Electric Co.
Type	Free Power Turbine
Reduction Gear Ratio	0.133
Prop Mfr.	Hamilton Standard
Prop. Dia.	84 in.
No. of Blades	3
Tail Pipe	Fixed Area

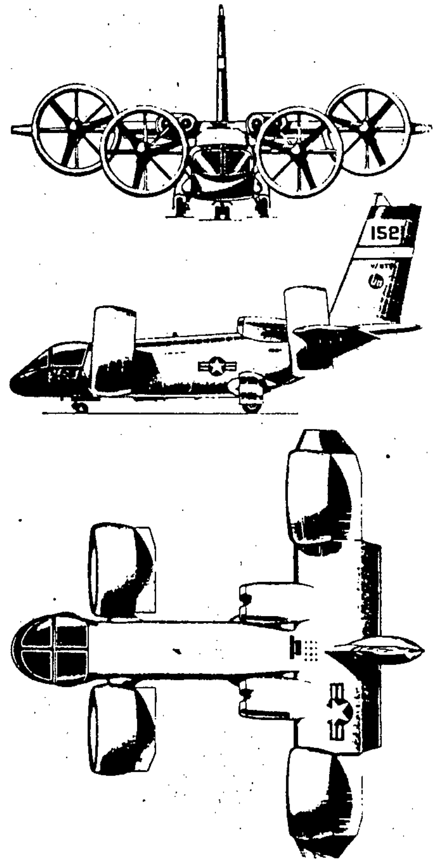
WEIGHTS

Loading	lb
Empty	11,622
Gross	15,287
Max Takeoff	18,420
Max Landing	15,287

FUEL

No. Tanks	Gal	Location
1	465	Fuselage

Fuel Grade JP-4 or JP-5



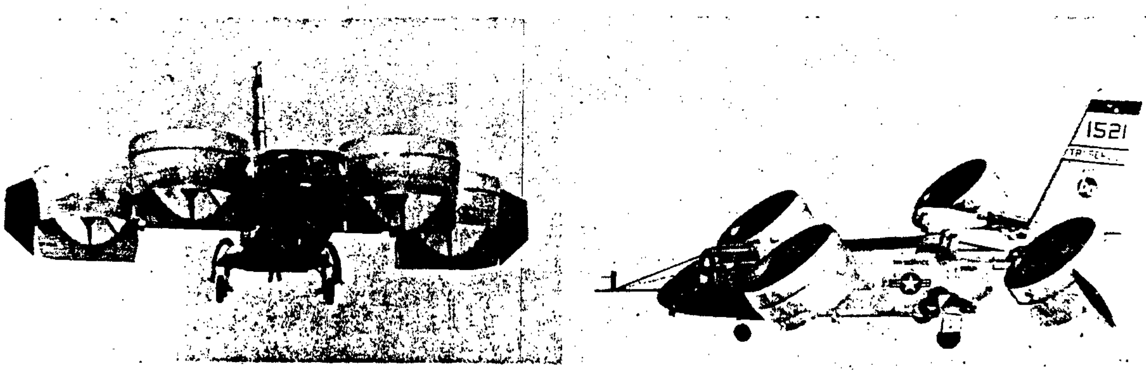
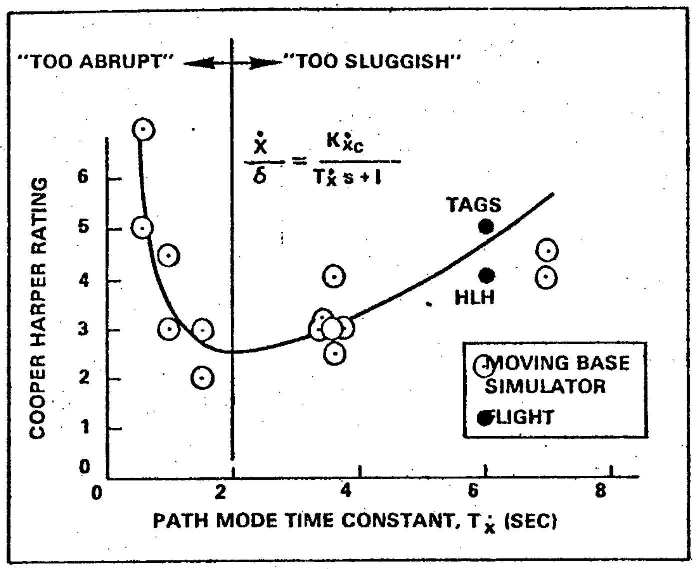


FIGURE 18. X-22A VARIABLE STABILITY V/STOL RESEARCH AIRCRAFT



$$\frac{\dot{x}}{\delta} = \frac{K_{\dot{x}_c}}{T_{\dot{x}}s+1}$$

FIGURE 19. TRANSLATIONAL RATE CONTROL RESPONSE REQUIREMENTS

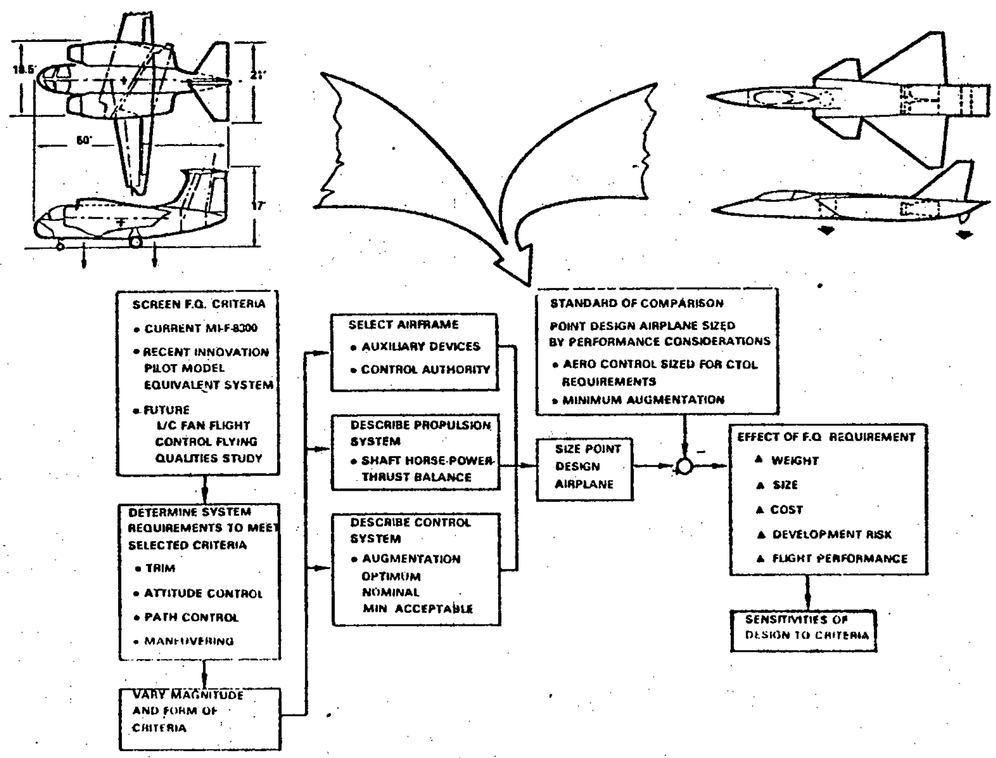


FIGURE 20. V/STOL DESIGN SENSITIVITY STUDY PLAN

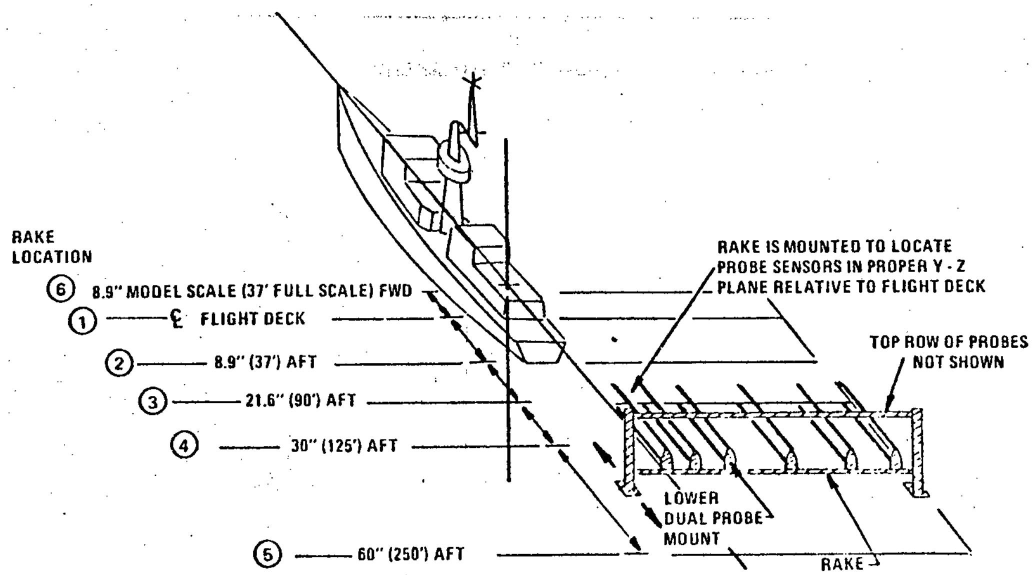


FIGURE 21. SMALL SHIP AIRWAKE TEST SETUP

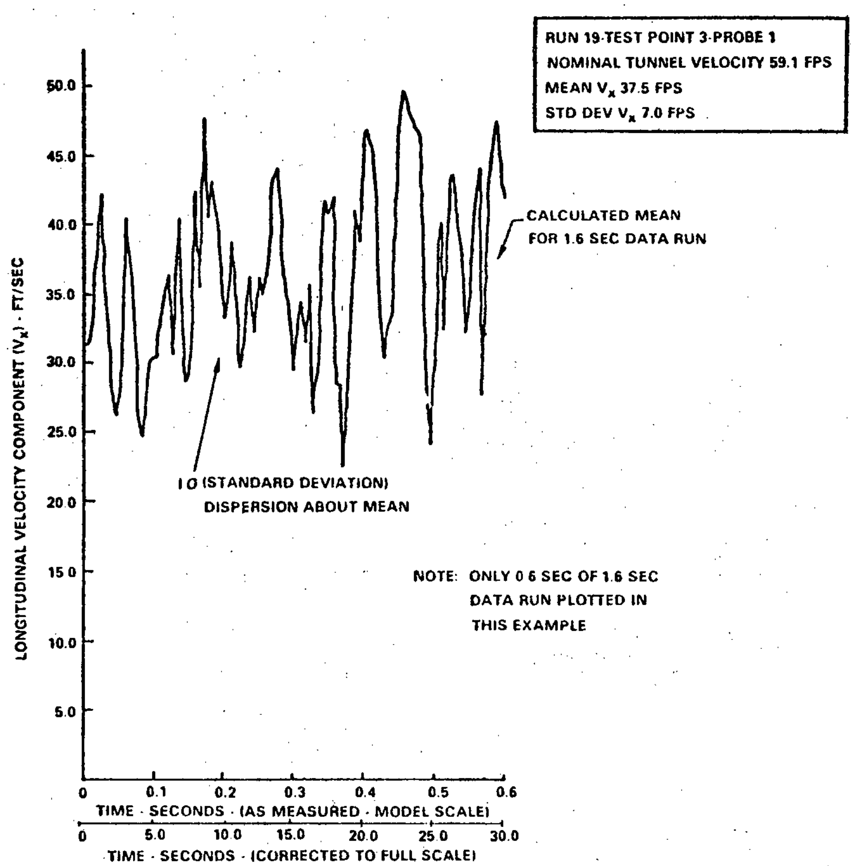


FIGURE 22. SAMPLE AIRWAKE TEST RESULTS

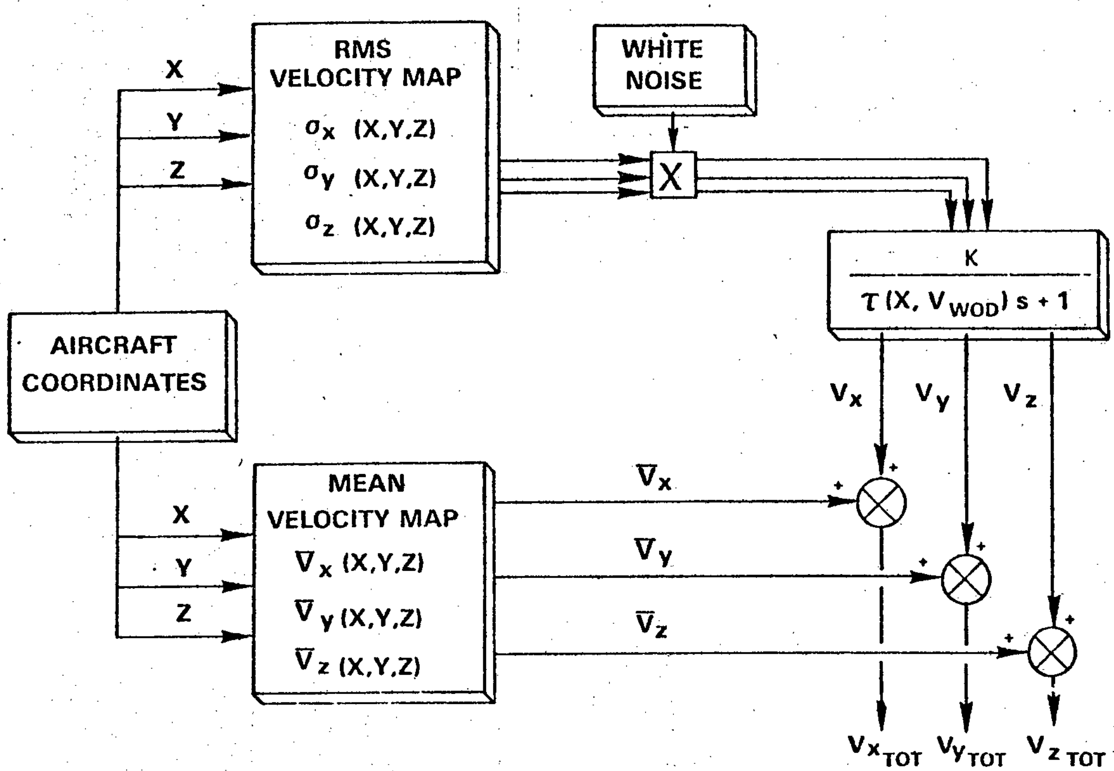


FIGURE 23. SMALL SHIP AIRWAKE MODEL

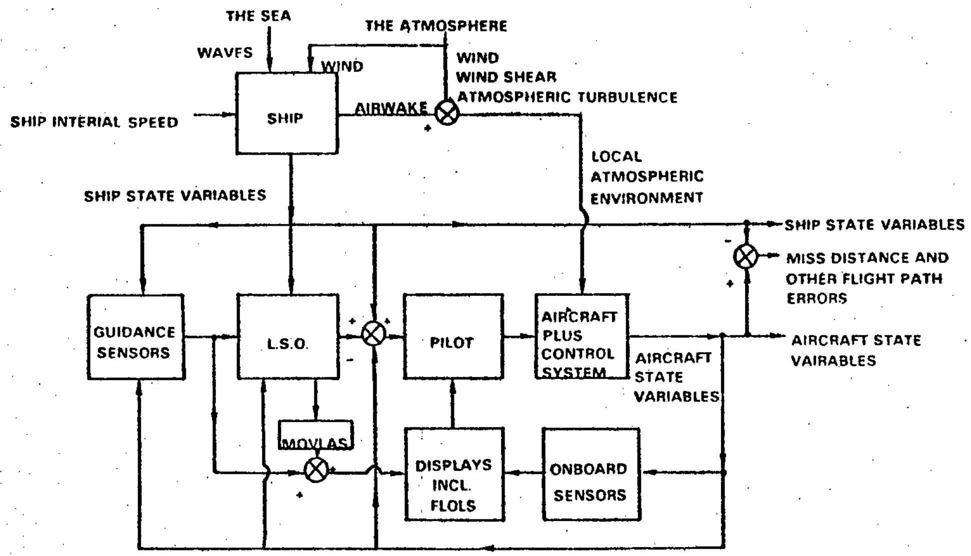


FIGURE 24. V/STOL LAUNCH AND RECOVERY DYNAMIC ELEMENTS

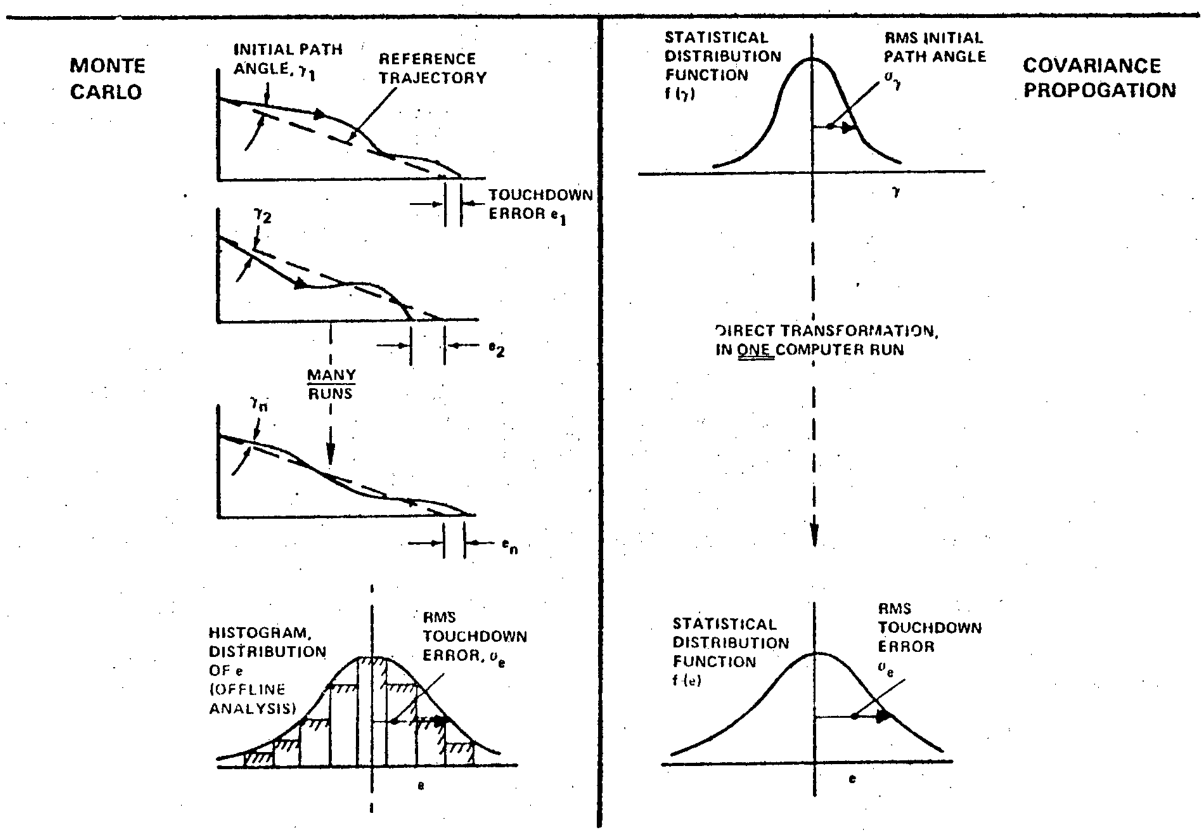


FIGURE 25. MONTE CARLO VS. COVARIANCE PROPOGATION METHODS

Biographical Sketch

John W. Clark, Jr., was born in West Reading, Pennsylvania on October 18, 1946. He received a B.S. degree, Magna Cum Laude, in Aerospace Engineering, from the Pennsylvania State University in 1968. He attended post-graduate studies at the Polytechnic Institute of Brooklyn leading to an M.S. degree in Aeronautics and Astronautics in 1972.

As an Aerospace Engineer employed by Grumman Aerospace Corporation, Mr. Clark performed design and development work in the areas of flight controls, flying qualities, and trajectory analysis on various Naval aircraft. He was actively involved in the F-14/Phoenix missile separation analysis and test program. Prior to leaving Grumman, he concentrated on the development of jet-lift V/STOL stability and control characteristics prediction techniques in hover and transition.

Mr. Clark is currently employed by the Naval Air Development Center, Warminster, Pennsylvania, as an Aerospace Engineer in the Flight Dynamics Branch. His responsibilities include stability, control and handling qualities evaluation of current and proposed Navy aircraft--both conventional and V/STOL.

Mr. Clark is a registered Professional Engineer and a member of Sigma Gamma Tau, Phi Kappa Phi, Sigma Tau, and the American Institute of Aeronautics and Astronautics (AIAA). He is a member of the Atmospheric Flight Mechanics Technical Committee of the AIAA.

Mr. Henderson was graduated from Princeton University in 1948 with a B.S. in Mechanical Engineering. He completed a number of graduate studies in Aeronautical Engineering at the University of Virginia and at UCLA while employed as a research engineer at NASA, Langley and as an aerodynamicist at North American Aviation, Downey, California. He gained extensive experience in V/STOL aerodynamics, stability and control at Bell Aerospace Company, Buffalo, New York as chief of aerodynamics during development of the X-22A and as Technical Director of several V/STOL design synthesis programs. He is presently responsible for technology developments in V/STOL aerodynamics at the Naval Air Development Center.

COST EFFECTIVE THRUST DRAG ACCOUNTING

BY

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Cost Effective Thrust Drag Accounting

Abstract

The government and private industry have in the past gone to a great deal of trouble and expense in quantifying additive drag. A thrust-drag accounting method has been developed which does not require that additive drag be quantified in order to obtain net thrust minus drag. By including inlet drag in the thrust, the omission of additive drag will occur in both thrust and drag and consequently the error will cancel when thrust and drag are combined. It is suggested that additive drag only be quantified when comparing wind-tunnel-derived drag with theoretical methods. A computer code was developed which will calculate additive drag for supersonic wing-bodies at angle of attack by coupling method of characteristics, geometric, and linearized drag prediction codes.

Basic momentum equations were applied to three thrust-drag accounting methods (free stream, inlet ramp, and inlet plane) for an air-breathing ramjet missile and determined in as rigorous a fashion as possible the best of the three, and the ground tests required to provide all necessary terms.

Introduction

For the past two years AEDC has been involved in obtaining ground test data and preparing in general for a wind tunnel flight correlation of the Advanced Strategic Air Launched Missile (ASALM). This preparation involved a study of various thrust drag accounting methods with particular emphasis on the treatment of additive drag, since for ASALM it is a significant percentage of the total drag. Unfortunately it was not felt that a consensus of opinion existed, either from the literature or in talking to individuals, as to what the role of additive drag should be in thrust-drag accounting or even whether it was a real drag or not. In the past the government and private industry have gone to a great deal of trouble and expense in quantifying additive drag and it was not apparent to us that it was necessary to quantify additive drag if you are concerned only with overall performance.

As a consequence of the apparent lack of consensus, we did a thrust-drag accounting study which primarily involved starting from the basic momentum equations. It involved deriving a thrust-drag accounting system which does not require that additive drag be quantified in order to obtain net thrust minus drag. The momentum equations were also applied to three thrust-drag accounting methods; the free stream, inlet ramp, and inlet plane and determined in as rigorous a fashion as possible the best of the three, and the ground tests required to provide all necessary terms. A computer code was developed which will calculate additive drag for supersonic wing-bodies at angle of attack by coupling method of characteristics, geometric, and linearized drag prediction codes.

Discussion

I would first of all like to briefly summarize in the first figure the content of the paper. There are, in general, two areas where I believe cost savings can be made in thrust-drag accounting: eliminating the quantification of additive drag when the interest is in obtaining performance of a given system, and selecting a thrust-drag accounting method that minimizes the requirement to define reference conditions which will usually result in a reduction of required ground testing. There are situations, however, when additive drag must be evaluated, for example, in comparing wind tunnel drag to theoretical prediction, and in comparing the drags of various inlets, or in estimating inlet drag.

I will derive from the basic momentum equation a method which omits additive drag in both the thrust and drag and which by including inlet drag in the thrust causes the "error" of omission which exists in both thrust and drag to cancel when thrust and drag are combined. I will briefly describe the additive drag prediction code and how the various pieces of it fit together; and I will give a summary of the results of the thrust-drag accounting method comparison with regard to advantages and disadvantages of each method, the ground tests required, and the applicability to theoretical prediction codes.

The sketch in the second figure defines the control volume of the captured air for a ramjet cruise missile in the supercritical supersonic mode. For the subsonic or for the supersonic subcritical case where the normal shock is outside, the basic formulation as presented here would apply as far as the momentum equation and the basic terms are concerned, although proper simulation of the flight vehicle in ground tests would be more difficult.

The control volume extends from free stream to the nozzle exit plane. "f" is the local friction force per unit area, and δ is the local slope and is positive when the local surface faces downstream (i.e., when the pressure force acts in a downstream direction). The directions in which the various forces act on the control volume are shown. Additive drag (or pre-entry force as it is often called) is defined in this paper as a pressure force acting on the captured stream surface and consequently increases the momentum of the captured air. The captured stream surface is that curvilinear surface which extends from the bow shock to the inlet lip and can be thought of as the surface generated by the streamlines which hit the inlet lip. Consequently, the additive drag should not be thought of as a real drag since it does not act on a real surface but rather as a pressure term in the momentum equation for the internal flow just as the free-stream pressure term ($\int p_\infty dA_\infty$) and the exit pressure term ($\int p_e dA_e$).

The sum of the pressure and friction forces acting on the fluid at the boundaries of the control volume is equal to the rate of change of momentum of the fluid flowing through the control volume as expressed in the following equation:

$$\int_{\infty} p_\infty dA_\infty + \int_{c.s.} p_{c.s.} \sin\delta \, dA_{c.s.} + \int_{w} (p_w \sin\delta - f\cos\delta)\, dA_w$$

$$+ \int_{r} (p_r \sin\delta - f\cos\delta)\, dA_r + \int_{int} (p \sin\delta - f\cos\delta)\, dA_{int}$$

$$+ \int_{e} p_e \sin\delta \, dA_e = \int_{e} \rho_e V_e^2 \cos^2 \phi_e \, dA_e - \dot{M}V_\infty \qquad (1)$$

where f is the local friction force per unit area and δ is the local surface angle. The first term in Equation (1) is the force that free-stream static pressure exerts on the control volume, the second term is the additive drag term or the force that the captured stream surface exerts on the control volume, the third and fourth terms represent the force that the forebody wedge and the inlet ramp exert on the control volume, the fifth term represents the internal force on the control volume, and the sixth term is the force that the exit static pressure exerts on the control volume. The sum of these forces is then equal to the rate of change of momentum as expressed on the right-hand side of the equation. In order to eliminate the A_∞ term, which is difficult to evaluate, Equation (1) is referenced to free-stream conditions by subtracting the identity

$$\int_{\infty} p_\infty \, dA_\infty + \int_{c.s.} p_\infty \sin\delta \, dA_{c.s.} + \int_{w} p_\infty \sin\delta \, dA_w$$

$$+ \int_{r} p_\infty \sin\delta \, dA_r + \int_{int} p_\infty \, dA_{int} + \int_{\infty} p_e \sin\delta \, dA_e = 0 \qquad (2)$$

from Equation (1). Equation (2) is an identity based on the fact that a constant pressure (p_∞) around a closed boundary (in this case the control volume) must have zero net force.

After subtracting the two equations the A_∞ term drops out and after transposing terms and writing the equation in more conventional terms, an expression for the propulsion drag is derived:

$$D_{prop} = D_w + D_r + D_{int} = \dot{M}V_\infty - F_e + D_{add} \qquad (3)$$

where the propulsion drag is expressed in terms of the wedge, ramp, and internal drags which are equal to the ram drag, nozzle exit stream thrust, and additive drags. The wedge

drag is the drag on the wedged-shaped portion of the forebody outlined by the dotted line which is scrubbed by the captured air. The particular terms included in the propulsion drag equation are dependent on the reference conditions used to separate internal and external drag. The equation as written here assumes the free-stream accounting method is used, that is, the inlet begins at free-stream conditions and includes the wedge and ramp. Consequently, the wedge and ramp drags are included in the propulsion drag. For an aircraft the wedge drag would be equivalent to the losses of the captured air caused by the fuselage forebody or wing or whatever affects the inlet flow. If the reference conditions are to be based on the inlet face plane, then D_{prop} would be equal to D_{int}. If the inlet ramp accounting method is used then $D_{prop} = D_r + D_{int}$ since the ramp is then considered part of the inlet. Regardless of the reference system chosen, however, quantifying the propulsion drag requires quantifying the exit momentum conditions and either the inlet reference conditions, the additive drag, or both. The wind tunnel aero-correction equation is given by Equation (4):

$$D_{corr} = D_{bal} - D_{base} - D_{prop} \tag{4}$$

where D_{bal} is the total balance measured drag, D_{base} is the base drag, and D_{prop} is the propulsion drag which again could include the wedge and/or ramp terms depending on the drag accounting method.

As written, Equation (4) assumes that the additive drag has been calculated and included in the propulsion drag calculation. Calculating additive drag, however, is a time-consuming, expensive, and relatively inaccurate practice.

If one elects not to go to the expense of evaluating additive drag, then the aerodynamic drag is expressed by Equation (5) where the drag is "in error" by an amount equal to the additive drag:

$$D = D_{corr} + D_{add} \tag{5}$$

The equation for thrust, Equation (6), is identical to that for propulsion drag except that the signs are changed:

$$T_{corr} = T_{int} - D_w - D_r = F_e - \dot{M}V_\infty - D_{add} \tag{6}$$

As written, the free-stream method is assumed for Equation (6) and only the exit stream thrust need be evaluated. It is inherently assumed, however, in this equation that the flow conditions at the propulsion test connect station (usually the compressor face) simulates the correct flow conditions. Again, if additive drag is not evaluated the thrust is "in error" by an amount equal to additive drag [Equation (7)]:

$$T = T_{corr} + D_{add} \tag{7}$$

Consequently, when net thrust is obtained, the two additive drag errors cancel and a true, corrected net thrust results:

$$T - D = T_{corr} + D_{add} - D_{corr} - D_{add} = T_{corr} - D_{corr} \tag{8}$$

Again it must be emphasized, however, that the aerodynamic and the thrust measurement tests must have identical flow conditions for this cancellation to be proper. Unfortunately, this is not typically the case for ramjet tests. It behooves us, therefore, to design models and plan tests that will provide for complementary simulation between aerodynamic and propulsion tests. If this simulation is provided I see no reason for going to the trouble of calculating additive drag when the primary interest is in obtaining system performance. The division of responsibility between airframers and propulsion people is somewhat arbitrary anyway, so there is no reason why both can't carry an extraneous additive drag term as long as it is agreed to by both parties. This is, in fact, being done by two private companies involved with developing a major Air Force system, although they won't admit that it is a quote, error, unquote, possibly because they are afraid the government will ask them to estimate the additive drag.

If in the course of the development of a system, performance falls below expectations and propulsion and airframers are pointing their fingers at each other, then the additive drag can be calculated for selected points only, which would be far better than calculating additive drag for the entire flight profile initially.

There are situations where additive drag must be calculated however; for example, it must be estimated for the comparison of wind-tunnel-measured drag and theoretically predicted drag. In this case the wind tunnel aerodynamic drag must be corrected for additive drag. Another use would be

in inlet design where a number of candidate designs must be compared. In general, additive drag must be evaluated any time the exit momentum must be corrected for the additive pressure term. For these situations where additive drag must be evaluated, it should be done as efficiently and accurately as possible. To that end we have developed a semi-automated procedure for calculating additive drag.

The procedure utilizes a number of existing computer codes as shown. At present the system is limited to axisymmetric bodies with circular lip inlets because the steamline calculator at present is an axisymmetric method of characteristics code, but we have plans for substituting a finite difference code which can handle arbitrary bodies at angle of attack. In these cases the captured stream surface would have a complex shape so the x-section generator should be especially useful. The x-sections are plotted and the areas are calculated. The area distribution is then fed into a potential flow drag prediction code. The additive drag can then be found by simply ratioing the inlet wrap angle to 360 degrees. In order to check the accuracy of this procedure, the additive drag was calculated for an axisymmetric inlet utilizing a right circular cone and was found to be within four percent of the exact solution of Sims.

Three thrust-drag accounting systems were compared for their applicability to ramjet cruise missiles, but the study was very general and should apply to any air-breathing vehicles. The three accounting systems differed only with regard to the reference conditions which divide internal and external aerodynamics. As described before, the three reference conditions studied were the free-stream method, i.e., the inlet begins at free stream, the inlet ramp method, and the inlet plane method. I will only give a summary of the results of the study. If more detail is desired, there is an AEDC technical memorandum report which can be consulted.

The free-stream method has the inherent advantage that no reference conditions need to be defined, whereas the other two methods require that the conditions at the inlet ramp shock and the inlet plane, respectively, be defined. The requirement to define reference conditions is an unnecessary expense when the known free-stream conditions could be used just as well. Defining reference conditions could also result in an inaccuracy as far as the division between thrust and drag is concerned because of the inherent difficulty of accurately defining the reference conditions. However, this inaccuracy would have no effect on thrust-drag. There is

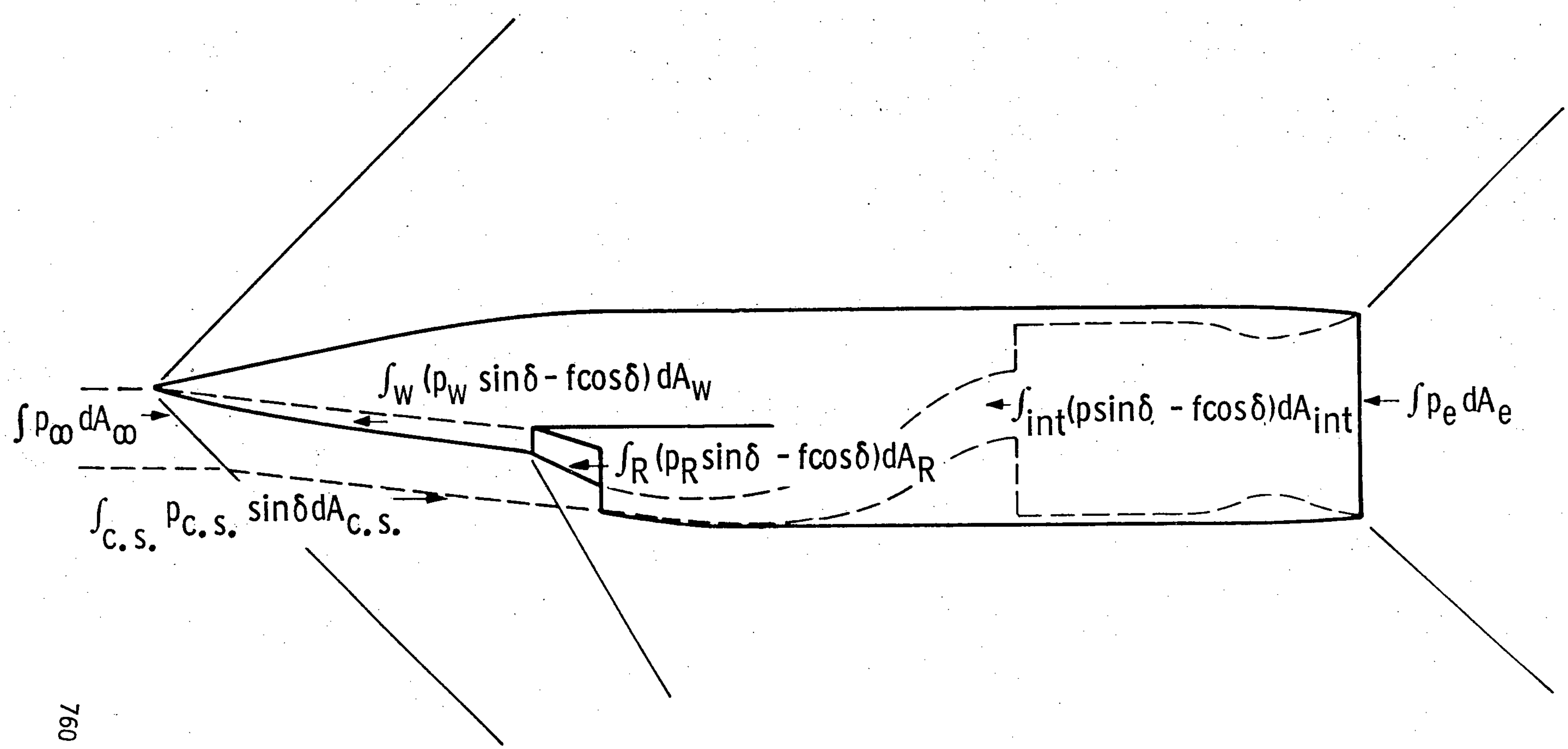
also the ever-present temptation to use expensive ground tests to define the reference conditions as a check to the cheaper but somewhat inaccurate theoretical prediction. Consequently, it appears that the free-stream method is the most cost-effective accounting method. Other than the possible requirement to define the reference conditions, the required ground tests are the same for all three methods. All three methods would require that conditions be defined at the propulsion test connect station, for example, at the compressor face for a jet engine or at the direct-connect station for a ramjet. Of course, for a free-jet test this requirement would be precluded.

Another comment concerning cost-effective drag accounting would be to admonish those involved with wind tunnel model design to keep the internal ducting as simple as possible, preferably a straight-through duct so that first, internal drag can be calculated and second, so that the internal drag will be kept low and consequently the internal drag error will hopefully be low.

SUMMARY

- ADDITIVE DRAG IN THRUST-DRAG ACCOUNTING
  - QUANTIFYING UNNECESSARY FOR OBTAINING NET THRUST MINUS DRAG
  - AUTOMATED PREDICTION METHOD FOR USE IN THEORETICAL COMPARISONS
- COMPARISON OF THREE THRUST-DRAG ACCOUNTING METHODS
  - ADVANTAGES AND DISADVANTAGES
  - GROUND TESTS REQUIRED
  - APPLICABILITY TO THEORETICAL PREDICTION CODES

INTERNAL DRAG MOMENTUM EQUATION TERMS



$$\int_{\infty} p_{\infty} dA_{\infty} + \int_{c.s.} p_{c.s.} \sin\delta \, dA_{c.s.} + \int_{w} (p_w \sin\delta - f\cos\delta)\, dA_w$$
$$+ \int_{r} (p_r \sin\delta - f\cos\delta)\, dA_r + \int_{int} (p \sin\delta - f\cos\delta)\, dA_{int}$$
$$+ \int_{e} p_e \sin\delta \, dA_e = \int_{e} \rho_e V_e^2 \cos^2\phi_e \, dA_e - \dot{M}V_{\infty} \qquad (1)$$

$$\int_{\infty} p_{\infty} dA_{\infty} + \int_{c.s.} p_{\infty} \sin\delta \, dA_{c.s.} + \int_{w} p_{\infty} \sin\delta \, dA_w$$
$$+ \int_{r} p_{\infty} \sin\delta \, dA_r + \int_{int} p_{\infty} dA_{int} + \int_{\infty} p_e \sin\delta \, dA_e = 0 \qquad (2)$$

PROPULSION DRAG

$$D_{prop} = D_w + D_r + D_{int} = \dot{M}V_\infty - F_e + D_{add} \tag{3}$$

AERO DRAG

$$D_{corr} = D_{bal} - D_{base} - D_{prop} \tag{4}$$

AERO DRAG UNCORRECTED FOR ADDITIVE DRAG

$$D = D_{corr} + D_{add} \tag{5}$$

THRUST

$$T_{corr} = T_{int} - D_w - D_r = F_e - \dot{M}V_\infty - D_{add} \tag{6}$$

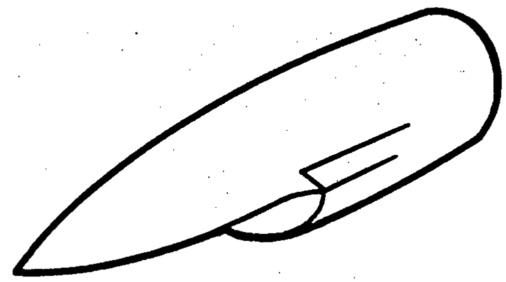
THRUST UNCORRECTED FOR ADDITIVE DRAG

$$T = T_{corr} + D_{add} \tag{7}$$

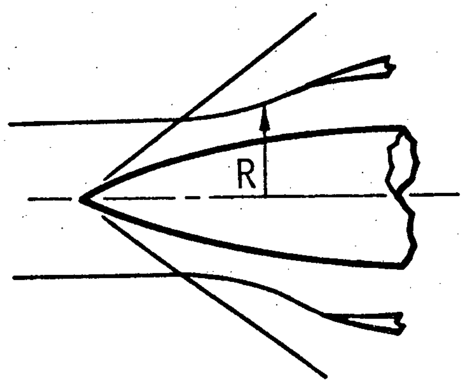
NET THRUST

$$T - D = T_{corr} + D_{add} - D_{corr} - D_{add} = T_{corr} - D_{corr} \tag{8}$$

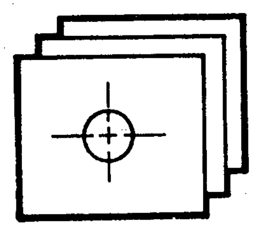
SEMI AUTOMATED ADDITIVE DRAG PREDICTION CODE



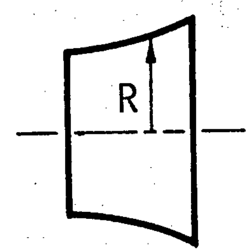
AXISYMMETRIC FOREBODY WITH
CIRCULAR LIP INLET



CALCULATES CAPTURED
STREAM SURFACE



GENERATES AND
PLOTS X-SECTIONS
OF STREAM SURFACE



CALCULATES AND RATIOS
POTENTIAL FLOW DRAG
OF STREAM SURFACE

SUMMARY OF DRAG ACCOUNTING METHODS

- FREE STREAM METHOD
  - ADDITIVE DRAG MUST BE EVALUATED OR ALLOWED TO CANCEL
  - NO REFERENCE CONDITIONS TO BE QUANTIFIED IN GROUND TESTS OR OTHERWISE
  - REQUIRES DEFINITION OF CONDITIONS AT PROPULSION TEST CONNECT STATION
- INLET RAMP METHOD
  - ADDITIVE DRAG MUST BE EVALUATED OR ALLOWED TO CANCEL
  - REFERENCE CONDITIONS MUST BE QUANTIFIED
  - REQUIRES DEFINITION OF CONDITIONS AT PROPULSION TEST CONNECT STATION
- INLET PLANE METHOD
  - NO ADDITIVE DRAG
  - REFERENCE CONDITIONS MUST BE QUANTIFIED
  - REQUIRES DEFINITION OF CONDITIONS AT PROPULSION TEST CONNECT STATION

Biographical Sketch

Russell B. Sorrells III was born in Atlanta, Georgia on March 15, 1938. He received a B.A. degree in Philosophy in 1959 from Emory University and a B.A.E. in 1962 from Georgia Tech.

He was commissioned in the U. S. Army through the Army R.O.T.C. program in 1962 and after six months training was assigned to NASA's Langley Research Center for the remainder of a two-year tour of duty. After being discharged he continued to work at Langley Research Center as a research engineer in the High Speed Aircraft Division/Supersonic Mechanics Section until 1975 and in the Aeronautical Systems Division/Vehicle Integration Branch until March 1976. Since that time he has been employed by the U. S. Air Force at Arnold Engineering Development Center in the Analysis and Evaluation Division.

DRAG PREDICTION FOR WING-BODY-NACELLE CONFIGURATIONS

BY

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Drag Prediction for Wing-Body-Nacelle Configurations*

Abstract

A theoretical method is being developed for predicting the aerodynamic drag of wing-body-nacelle configurations at subsonic and supersonic speeds. A modified axisymmetric analogue is employed in determining the turbulent skin friction of the subject three-dimensional bodies. An integral axisymmetric boundary layer technique is applied along streamline and longitudinal boundary layer flow properties are obtained. The cross-flow quantities are first assumed as constants and then updated with calculated longitudinal values. In order to implement the modified axisymmetric analogue, an exact method for determining the streamline pattern and transverse scale factors over general three-dimensional bodies has been developed. The required surface pressures are obtained by using a three-dimensional potential flow program. Good comparison between the calculated drag value and experimental measurement was obtained for the test case of a wing-body configuration at $M_\infty = 0.4$.

In the supersonic case, drag predictions were performed for a supersonic V/STOL configuration with various store arrangements. External stores included short- and medium-range missiles, electronic pod, and 600-gallon (2280 liters) fuel tanks. Predictions were based on cruise conditions at Mach numbers 1.2, 1.6, and 1.8 with Reynolds numbers between 2×10^6 and 66.2×10^6. The wave drag varies non-linearly with added store capacity. Data indicate that an optimum arrangement of external stores can be found to minimize the supersonic wave drag.

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I. Introduction

The drag of a vehicle is directly related to the thrust requirement (T=D) and energy consumption (P=DV) and, therefore, its precise determination is of vital importance to the vehicle's performance and maneuverability. Presently, there exist a number of computer codes for determining lift and other aerodynamic characteristics of three-dimensional bodies, but none provides capabilities for drag prediction. The reason for the lack of such capabilities lies in the complexity of handling the three-dimensional boundary-layer flows. Methods for calculating three-dimensional boundary layers have not been developed to the state that they can be applied to a complete vechile configuration. Sophisticated numerical solutions are costly and restricted to simple geometries or some component of a vehicle (for instance, see References 1 and 2).

In the present work, a computational method for predicting three-dimensional drag of a complete aircraft configuration in subsonic and supersonic flows is being developed. The method employs a surface streamline system and the turbulent skin friction coefficient along a streamline is determined. Exact equations for calculating the surface streamlines over arbitrary wing and fuselage shapes were derived. The present approach has the simplicity of small cross-flow approach[3,4] yet still retains three-dimensional features by incorporating the divergence of streamline and accounts for the cross-flow effect in the viscous solutions.

A drag prediction analysis was performed in the supersonic case to define the geometry and performance of a supercruiser concept for vertical and short takeoff and landing (V/STOL) to carry weapons compatible with the deck launched intercept and strike missions. Drag coefficients of a supersonic V/STOL configuration with seven store arrangements were determined. Theoretical drag consists of supersonic wave drag and turbulent skin friction. Computations were made to substantiate the data base of the supersonic phase of the present work.

II. Three-Dimensional Viscous Drag

A. Three-Dimensional Boundary-Layer Approximations

The aerodynamic drag of a wing-body-nacelle combination (see Figure 1) includes viscous drag and wave drag at transonic and supersonic speeds. In calculating the viscous drag, a system of orthogonal, curvilinear, coordinates (ξ,η,ζ) is used. As shown in Figure 2, the coordinate direction ξ coincides with the local external inviscid streamline projected in the plane tangent to the surface; η is also in

the tangent plane and normal to ξ; and ζ is measured from the surface along a straight line normal to the tangent plane. The metric is

$$dL^2 = (h_1 d\xi)^2 + (h_2 d\eta)^2 + d\zeta^2 \tag{1}$$

where $h_1 d\xi$ = ds, the length element along a streamline in the boundary layer

$h_1(\xi,\eta)$ = the scale factor for the ξ direction

$h_2(\xi,\eta)$ = the scale factor for the η direction

$h_3 = 1$

In this coordinate system, the equations governing the three-dimensional turbulent boundary layers of a homogeneous gas, in the absence of body forces and heat sources, may be written as

Continuity

$$\frac{\partial}{\partial\xi}(\rho h_2 u) + \frac{\partial}{\partial\eta}(\rho h_1 v) + \frac{\partial}{\partial\zeta}(\rho h_1 h_2 w) = 0 \tag{2}$$

ξ-Momentum

$$\rho\left[\frac{u}{h_1}\frac{\partial u}{\partial\xi} + \frac{v}{h_2}\frac{\partial u}{\partial\eta} + w\frac{\partial u}{\partial\zeta} - K_2 uv + K_1 v^2\right] = -\frac{1}{h_1}\frac{\partial P}{\partial\xi} + \frac{\partial}{\partial\zeta}\left(\mu_1\frac{\partial u}{\partial\zeta}\right) \tag{3}$$

η-Momentum

$$\rho\left[\frac{u}{h_1}\frac{\partial v}{\partial\xi} + \frac{v}{h_2}\frac{\partial v}{\partial\eta} + w\frac{\partial v}{\partial\zeta} - K_1 uv + K_2 u^2\right] = -\frac{1}{h_2}\frac{\partial P}{\partial\eta} + \frac{\partial}{\partial\zeta}\left(\mu_2\frac{\partial v}{\partial\zeta}\right) \tag{4}$$

ζ-Momentum

$$\frac{\partial P}{\partial\zeta} = 0 \tag{5}$$

where

$$K_1 = -\frac{1}{h_1 h_2}\frac{\partial h_2}{\partial\xi}, \quad K_2 = -\frac{1}{h_1 h_2}\frac{\partial h_1}{\partial\eta}$$

$$\mu_1 = \mu - \rho\,\overline{u'w'}/\frac{\partial u}{\partial\zeta}$$

$$\mu_2 = \mu - \rho\,\overline{v'w'}/\frac{\partial v}{\partial\zeta}$$

The other symbols have their usual meaning.

B. The Axisymmetric Analogue

In the solution procedure for the three-dimensional boundary-layer flows, great simplification may be achieved by assuming the cross-wise velocities and gradients to be small. Consequently, the products of these terms can be dropped in Equations (2) through (4), which, considering the streamwise flow, yields[5]

Continuity

$$\frac{1}{r}\frac{\partial}{\partial s}(\rho r u) + \frac{\partial}{\partial \zeta}(\rho w) = 0 \tag{6}$$

ξ-Momentum

$$\rho\left(u\frac{\partial u}{\partial s} + w\frac{\partial u}{\partial \zeta}\right) = \rho_e u_e \frac{\partial u_e}{\partial s} + \frac{\partial}{\partial \zeta}\left(\mu_1 \frac{\partial u}{\partial \zeta}\right) \tag{7}$$

ζ-Momentum

$$\frac{\partial P}{\partial \zeta} = 0$$

where

$$\frac{\partial}{\partial s} = \frac{1}{h_1}\frac{\partial}{\partial \xi}, \quad r = h_2$$

Equations (5), (6), and (7) are identical to those governing the axisymmetric turbulent boundary layers with the radius r of the spurious axisymmetric body and, therefore, can be solved by standard procedures for axisymmetric flows independently of the cross flow. The three-dimensional boundary-layer flow is, thus, represented by a number of axisymmetric flows; see Figure 3. The coordinate directions, however, must be taken along and perpendicular to the inviscid surface streamlines.

C. Modified Axisymmetric Analogue

For general three-dimensional bodies having large geodesic curvatures, such as the wing-body junction of an aircraft, or the bilge or afterbody of a ship, the cross flows inside the boundary layers play an important role in the resulting flow characteristics. We, therefore, have to retain the cross-flow velocity terms governed by the simplified momentum equation in the η direction:

$$\rho\left(u\frac{\partial v}{\partial s}+w\frac{\partial v}{\partial \zeta}+\frac{uv}{r}\frac{\partial r}{\partial s}\right) = K_2\left(\rho_e u_e^{\ 2}-\rho u^2\right)+\frac{\partial}{\partial \zeta}\left(\mu 2\frac{\partial v}{\partial \zeta}\right) \tag{8}$$

This equation is linear in v with u and w as parameters which are obtained separately from the solution to Equations (5) through (7).

The η derivatives would have to be determined by solving the full equations, Equations (2) through (4), with the resultant u, v, and w as constant parameters. These values are then updated with recalculated longitudinal flow properties and the whole solution proceudre is repeated until the changes in values of the crosswise terms are within a prescribed limit.

Again, the streamline geometry would have to be known in applying the modified axisymmetric analogue.

D. Skin Friction

The viscous drag is determined by the shearing stress in streamwise and crosswise directions as given by the following expressions:

$$\tau_1 = \mu\frac{\partial u}{\partial \zeta} - \rho\,\overline{u'w'} \tag{9}$$

$$\tau_2 = \mu\frac{\partial v}{\partial \zeta} - \rho\,\overline{v'w'} \tag{10}$$

Values of τ_1 and τ_2 are obtained, therefore, from the solution for u, v, and the eddy models for $\rho\,\overline{u'w'}$ and $\rho\,\overline{v'w'}$.

E. Solution Procedure

The integral approach to solving the equations should be adequate to yield the desired accuracy in a minimum of computer time. In so doing, Equations (7) and (8), and ultimately Equations (3) and (4), may be integrated term by term across the boundary layer using Equations (6) or (2) to eliminate w.

To perform the integration of streamwise flow, it is assumed that the velocity profile in the streamwise plane is similar to that for two-dimensional flow. Therefore, we may use the expression for the velocity profile given in Reference 6 which produced satisfactory results for attached turbulent boundary layers.

The crosswise velocity profile is related to the streamwise flow by Mager's relation:[7]

$$v = u\left(1-\frac{\zeta}{\delta}\right)^2 \tan\lambda \tag{11}$$

where λ is the angle between surface shearing stress and surface inviscid streamline, and is defined as

$$\tan\lambda = \lim_{\zeta\to 0} \frac{v}{u} = \frac{\tau_2}{\tau_1}$$

The boundary conditions on v are satisfied both at the wall and at the edge of the boundary layer.

The eddy viscosity model for streamwise flow used in the present work is similar to that of Kuhn and Nielsen,[8] and has been employed by the author previously.[6] For crosswise flow, a plausible formula may be obtained by replacing u terms with v which is subsequently determined by using Equation (11).

F. A Simplified Solution

To obtain the frictional drag in streamwise planes, an alternate approach to the above integral method is to consider a transformation of the classical flat plate solution to compressible axisymmetric flow with pressure gradient. A distinct advantage is its simplicity and reasonable accuracy for engineering purposes.

The transformation has been performed by the present author elsewhere.[9] In particular, after some manipulation of Equation (22) of Reference 9, the expression for turbulent skin friction coefficients is

$$C_f = \frac{0.0592\,\mu_e^{0.8}(\rho_e\nu_r)^{0.5}\,h_2^{0.25}}{\mu_o^{0.6}\left[\int_o^s \rho_e^{1.25}u_e\nu_r^{0.25}h_2^{1.25}\,ds\right]^{0.2}} \tag{12}$$

where the variables with subscript r are evaluated at reference conditions based on temperature.

Equation (12) is, therefore, readily applicable along a streamline with known flow properties at the edge of the boundary layer and the scale factor h_2. The three-dimensional effect is represented by

h_2 for divergence and convergence of streamlines, although no crossflow is accounted for in this simplified approach.

III. Calculation of Streamlines and Scale Factors

The success in applying the axisymmetric analogue (II-B), the modified axisymmetric analogue (II-C), or the simplified approach (II-F) along streamwise planes depends heavily on the ability to calculate streamlines and scale factors h_2.

The derivation of equations for the required streamline geometry starts with the full inviscid momentum equations over the surface of a general three-dimensional body. The geometric quantities of the body surface contained in momentum equations are specified by metric tensors g_{ij} in terms of coordinates x, y, and z, where x and y are measured along the body surface in longitudinal and transverse directions, respectively, and z is measured normal to the surface. For a wing, the longitudinal direction would be the chordwise direction; for a fuselage and a nacelle, it would be a meridian. The coordinate y is replaced by an azimuthal angle, for convenience, in the case of a fuselage or a nacelle. These coordinates, along with the basic Cartesian coordinates $(\overline{x},\overline{y},\overline{z})$ and the resulting streamwise coordinates (ξ,η,ζ) for wing, fuselage, and nacelle, are shown in Figure 4.

The procedure is essentially an extension of an earlier work formulated for axisymmetric bodies at various angles of attack.[9] In the present work, the coordinates (x,y,z) for the wing, or (x,ϕ,z) for a fuselage, are nonorthogonal, and, therefore, introduce complexities. The entire procedure is too lengthy to be presented in this paper. Rather, it will be documented in a DTNSRDC report in the near future.

IV. Determination of Inviscid Pressures

The inviscid pressures required by both solutions for the turbulent boundary layers and the streamline goemetry can be obtained either by theoretical calculations or by experimental measurements.

For low-speed subsonic flows, the three-dimensional potential flow solution developed by Hess[10] has been employed for determining inviscid pressures over the subject wing-body-nacelle configuration. The method is "exact" in the sense that the full three-dimensional potential flow equations were used without any approximations except those for overcoming some numerical realities. The Hess code has been accepted by other investigators in handling a complex configuration with reasonable accuracy.

Experimentally measured pressures can also be incorporated through proper interpolation schemes over three-dimensional body surfaces. In particular, a fourth-order Lagrangian interpolation formula proves to be convenient in representing continuous pressure distributions with relatively sparse data. The pressures and their derivatives are evaluated accordingly. Use of experimental pressures is especially desirable in the case of compressible flows where a theoretical inviscid flow solution is generally not available for complex configurations. Because the source of the pressure distribution has direct impact on the resulting skin friction values, the use of experimental pressures may also serve as test cases for either incompressible or compressible flow conditions.

V. Drag in Supersonic Flight

The theoretical drag of an aircraft in supersonic flight consists of the skin friction and the supersonic wave drag. Pressure drag due to flow separation is not considered.

A. Skin Friction

The method described in the previous section is good for determining the skin friction in a supersonic flow with valid pressure distributions.

For supersonic configurations with thin wing and slender fuselage, an alternative but crucial approach in predicting the skin friction is to apply a turbulent flat plate solution to local longitudinal strips alined with the free stream. The approach is called the T method which is based on the calculation of a compressible skin friction coefficient C_f from a reference skin friction coefficient C_f^* for a given free-stream Mach number, Reynolds number, and adiabatic wall temperature. The details of the method are given in Reference 11. In the solution procedure, the whole configuration is divided into many strips; the wetted area of these strips are then calculated. The total frictional drag of the configuration is determined by summing the strip values along with the corresponding wetted areas.

B. Supersonic Wave Drag

The supersonic wave drag was calculated by a far-field solution based on the supersonic area rule. A near-field solution based on the linear theory was used as a check in some cases.

1. The Far-Field Solution

In using the supersonic area rule, the aircraft configuration is transformed into several equivalent bodies of revolution by passing a series of parallel cutting planes through the configuration. The cutting planes are inclined with respect to the aircraft longitudinal axis at the Mach angle, and a single equivalent body is produced for the series of cutting planes at a constant azimuthal angle. The cross-sectional area of the equivalent body at each station is the projection of the area intercepted by the cutting plane onto a plane normal to the aircraft axis.

The wave drag of each equivalent body of revolution is then determined by von Kármán's slender-body theory.

2. The Near-Field Solution

The accuracy of the far-field solution was first checked by a more sophisticated near-field solution which gives more detailed information such as surface pressure distribution and interference effects. The entire solution involves calculations of "isolated" wave drag of the components and superposition of interference drag between the components.

In calculating the "isolated" wave drag, the surface pressure coefficient on the upper (or lower) surface of a flat-mean-line wing of symmetrical surface shape is obtained by first determining the corresponding velocity potential, differentiating with respect to x (to get u), and then computing the pressure coefficient from the approximation, $C_p = -2u/V_\infty$.

For fuselage and nacelles, the surface pressure distribution is obtained from a method based on the Lighthill theory. The method is applicable to bodies having either smooth area distributions or bodies with slope discontinuities. In computation, smooth area distributions are assumed except (if required) at the nose or aft end of the body. Open-nose bodies, such as nacelles, are permissible. The solution technique requires calculating an axial perturbation velocity which is a function of the body cross-sectional area growth (and radius distribution) and a decay function which relates area growth to its effect on a given field point.

The interference drag is accounted for by considering the near-field pressure signature transmitted among components of the whole configuration. Types of interference include fuselage-on-wing, wing-on-fuselage, nacelle-on-wing, wing-on-nacelle, nacelle-on-fuselage, and fuselage-on-nacelle; for details see Reference 11.

C. The Computer Program

The computer program used for the supersonic drag calculations is basically the supersonic aircraft analysis program described by Middleton et al.[11] with minor modifications made at DTNSRDC. Program options include the SKIN FRICTION, FAR-FIELD WAVE DRAG, NEAR-FIELD WAVE DRAG, WING DESIGN, LIFT ANALYSIS, and PLOT programs. Except for the WING DESIGN and LIFT ANALYSIS options, the program is fully utilized in this analysis. The PLOT routines were modified to adapt to the CALCOMP 936 plotting facility at DTNSRDC.

VI. Results and Discussion

To validate the method for predicting the three-dimensional drag over wing-body-nacelle configurations, a test case was carried out in subsonic flight and several sample calculations were performed in supersonic flight.

A. Test Case in Subsonic Flight

The wing had cross sections of NACA 65A-006, with an area aspect ratio of 5.86 and a sweep angle of 35 degrees. The fuselage was a pointed body of revolution with a fineness ratio of 12. This particular configuration was chosen because of its simplicity in shape and availability of experimental data for comparison purposes.

Using the experimental pressures of Reference 12, the streamline geometry over the fuselage and the wing was determined in accordance with the newly derived method. Because of symmetry, only one side was considered. A total of 31 streamlines over one side of the wing-body combination were calculated. The turbulent skin friction coefficients along the streamlines were determined and subsequently summed over the entire surface. The calculated total drag coefficient at $M_\infty = 0.4$ is $C_D = 0.008013$ which compares well with the measured value of $C_D = 0.0080$ taken from Reference 12. Theoretical pressures were also obtained by the Hess three-dimensional potential flow program with a total of 132 surface panels for one side of the wing-body combination. The difference in drag values by using theoretical and experimental pressures has not been evaluated at the time of this writing.

The drag coefficient for a complete wing-body-nacelle configuration was also calculated using a theoretical surface pressure determined by the Hess code with 204 panels. The nacelles contributed more than 20 percent of the drag increase, compared to that of the wing-body along as shown in Figure 5.

B. Drag of a Supersonic V/STOL Configuration

Drag coefficients of a supersonic V/STOL configuration were evaluated in order to substantiate the data base of the supersonic phase of the present work. An isometric view of the aircraft configuration is shown in Figure 6a and its munerical representation is given in Figure 6b. In preparing the inputs for the computer, the aircraft configuration surface is approximated by a large number of quadrilateral shapes defined by input coordinate points in physical space. The mission oriented stores (missiles) were converted to equivalent bodies of revolution and the radius determined at longitudinal stations for computer input data as illustrated in Figure 7.

The calculated results of the test case (NASA Supercruiser 4; see Figure 8) are plotted in Figure 8 along with independently calculated data and experimental measurements by Shrout.[13] Skin friction, wave drag, and drag due to camber are considered. Generally, the present calculated results compare very well with the experimental values at $M_\infty = 1.6$ and 1.8, but yield higher values at $M_\infty = 1.2$. This is expected because the linear theory starts to fall apart as the free-stream Mach number approaches unity. However, the accuracy of the linear theory may be improved to a certain extent by using a fine integration grid, as done by Shrout to yield better comparison at $M_\infty = 1.2$. Figure 8 also shows some drag values at subsonic speeds. Because the theory does not include induced drag, the subsonic results are considerably lower than actual measured values.

The drag coefficients were determined for the supersonic V/STOL configuration with seven store arrangements after the NASA Supercruiser 4 test case. Wave drag values were calculated at $M_\infty = 1.2$, 1.6, and 1.8, and skin friction values were determined at the same Mach numbers with corresponding Reynolds numbers. The detailed results were presented in Reference 14 while some typical cases are summarized below.

Typical cases are represented by four configurations as shown in Figure 9. Figure 10 depicts oblique orthographic views of Configuration 4. For the clean aircraft configuration, Configuration 1, both far-field and near-field wave drag values were determined and the difference between the two solutions was examined. Configuration 1 has larger total wetted area and, therefore, yields higher values of wave drag as well as skin friction. For Configurations 2 through 4, the wave drag coefficients were all determined by the far-field solution which were then converted to the near-field values based on correlation between the coefficients obtained by the two solutions for Configuration 1. The total drag coefficients were obtained by adding the skin friction. The total drag so determined, therefore, should include the interference drag and the drag due to camber, although they are not directly determined from the far-field solution.

Finally, Figure 11 depicts the variation of total drag coefficients versus free-stream Mach numbers. As expected, the drag coefficient decreases as the free-stream Mach number increases. The wave drag varies nonlinearly with added store capacity. This variation may be attributed to the interference of shock waves and to the difference in resulting equivalent bodies of revolution in accordance with the supersonic area rule. This difference implies that an optimum arrangement of external stores can be found to yield an equivalent body of revolution having the least supersonic wave drag. For instance, at $M_\infty = 1.8$ the drag value of the aircraft with four medium-range air-to-air missiles (Configuration 3) is 0.01102 and the drag value of the aircraft with four identical missiles plus two short-range air-to-air missiles (Configuration 4) is 0.01099. The wave drag decreases when two short range missiles were added.

VII. Conclusion and Future Efforts

The methodology now being developed for predicting three-dimensional drag of a general wing-body-nacelle configuration seems to yield reasonably accurate results for the preliminary design and performance evaluation of future naval aircraft. Data indicate that an optimum arrangement of external stores can minimize the supersonic wave drag.

Future efforts under the present task should include the following capabilities:

1. Prediction of drag with lift.
2. Prediction of drag of V/STOL configuration during transition mode with influence of lifting jets (see Figure 12).
3. Extension to transonic flight conditions.

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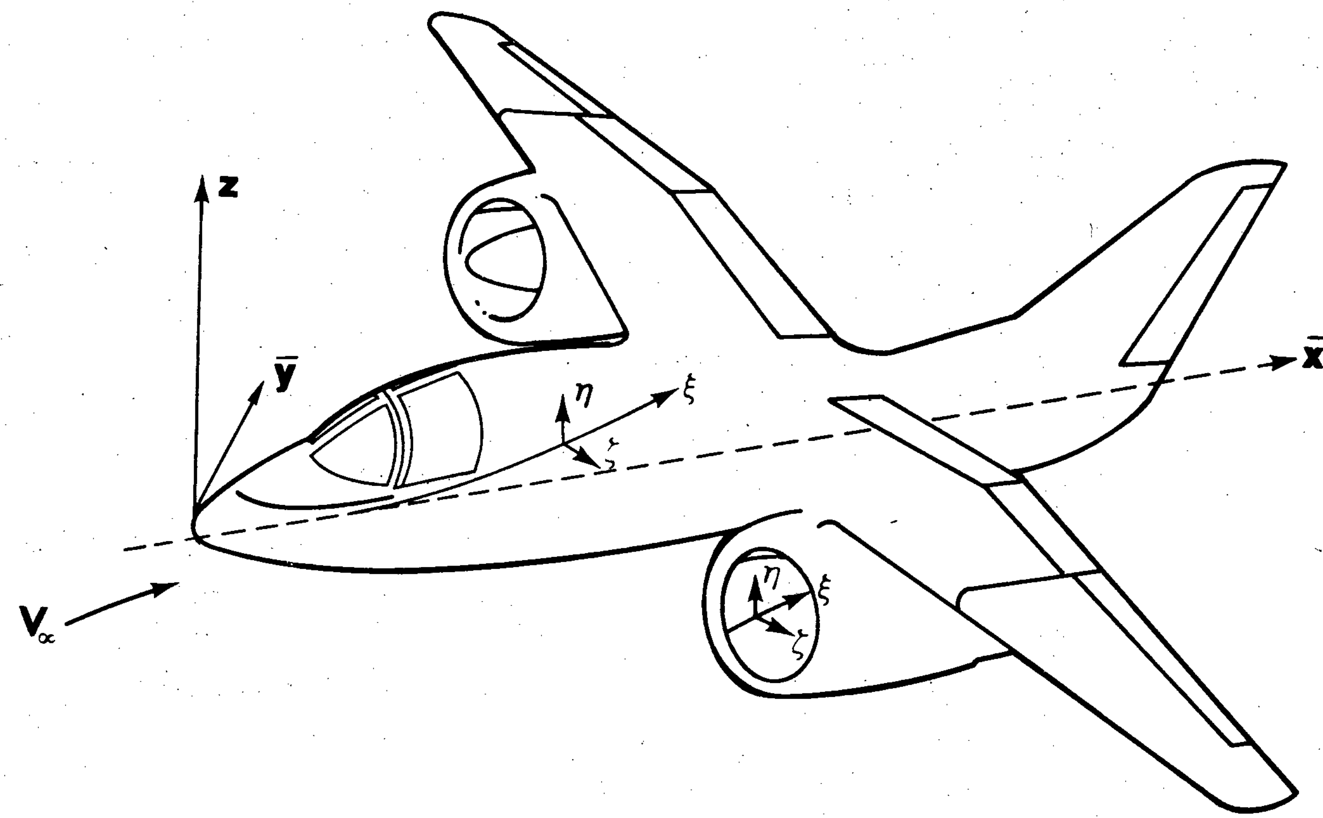
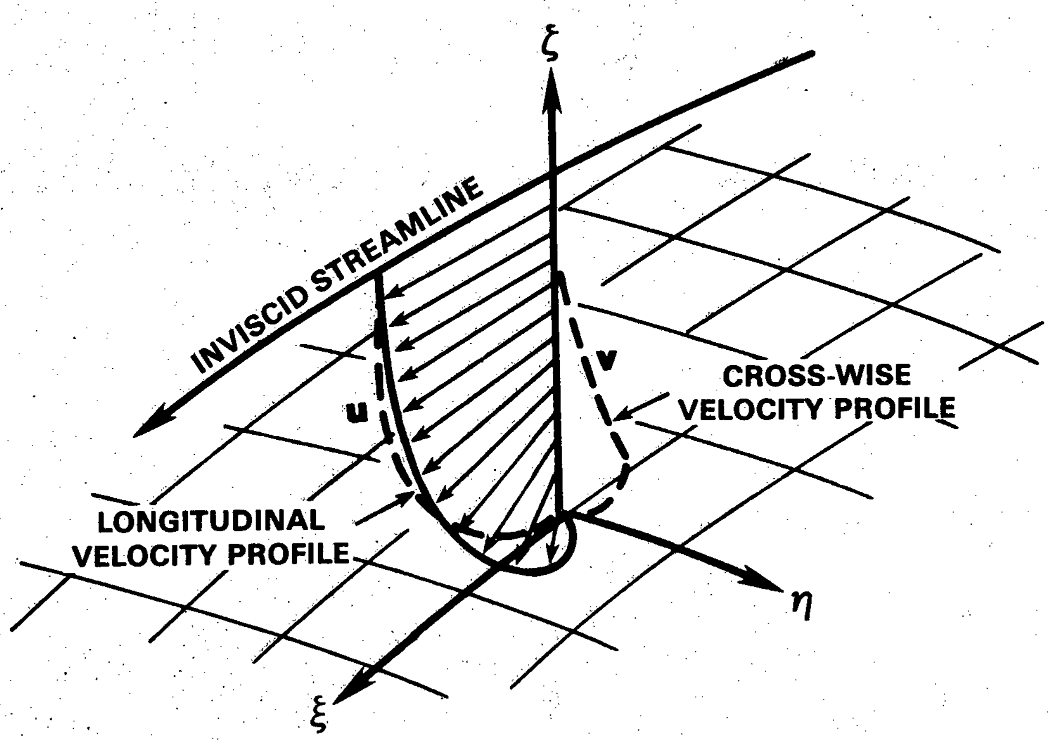


Figure 1 - Schematic of Wing-Body-Nacelle Configuration



BY ASSUMING v→0, THREE-DIMENSIONAL
BOUNDARY LAYER EQUATIONS REDUCE
TO AXISYMMETRIC FORM IN STREAMWISE SYSTEM

Figure 2 - Three-Dimensional Boundary Layer in a Streamwise Plane

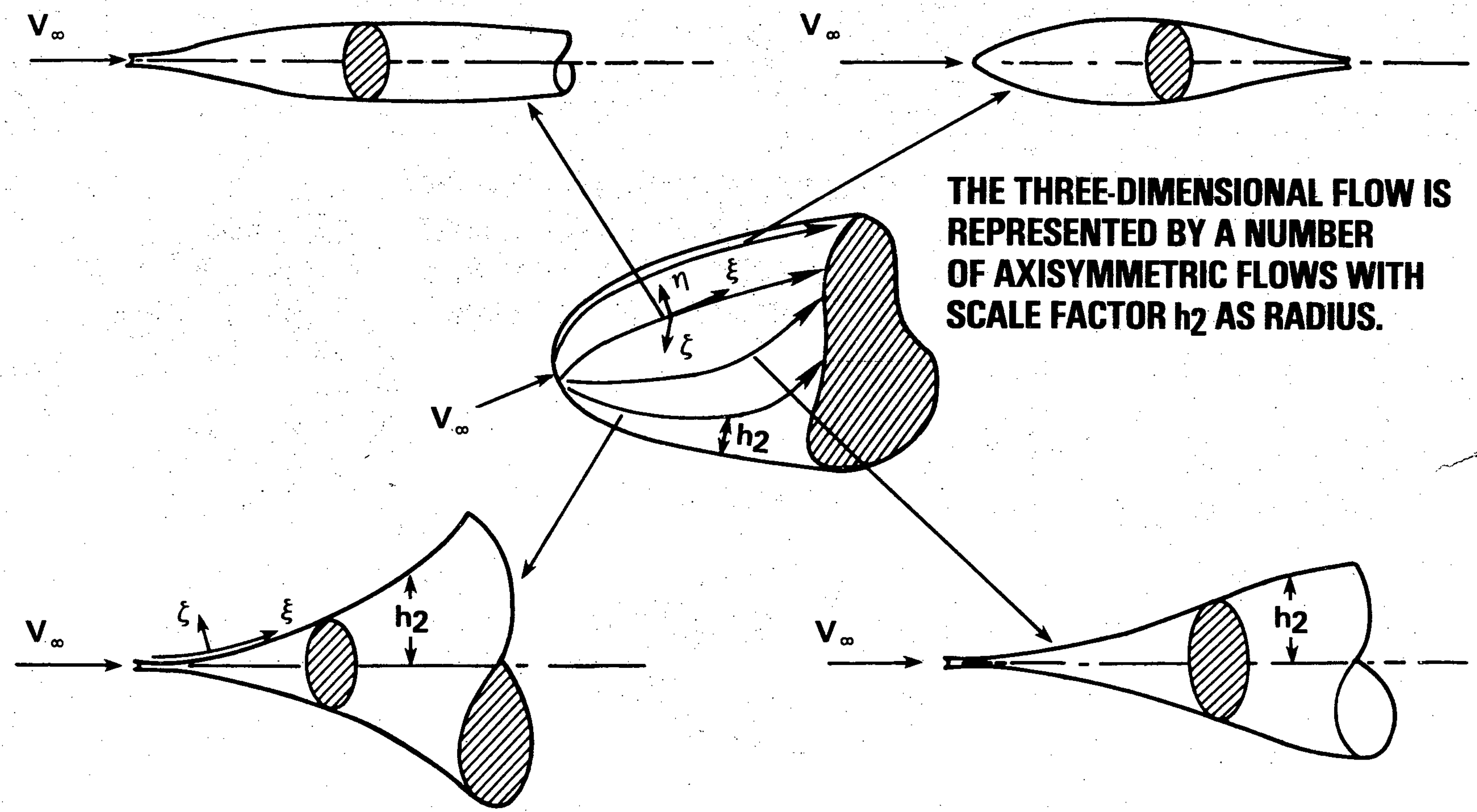


Figure 3 - The Axisymmetric Analogue

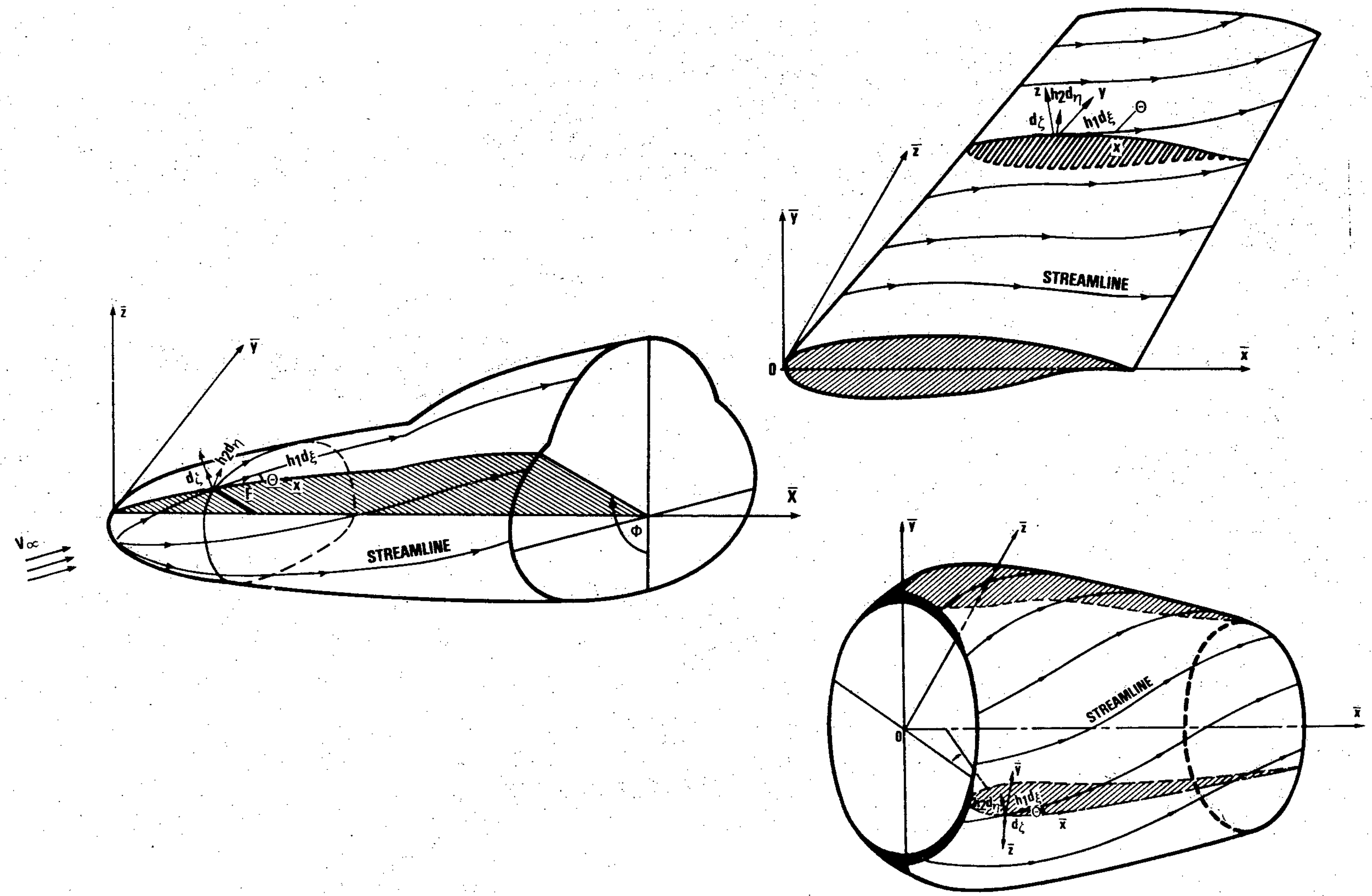


Figure 4 - Coordinate Systems for Components of a Wing-Body-Nacelle Configuration

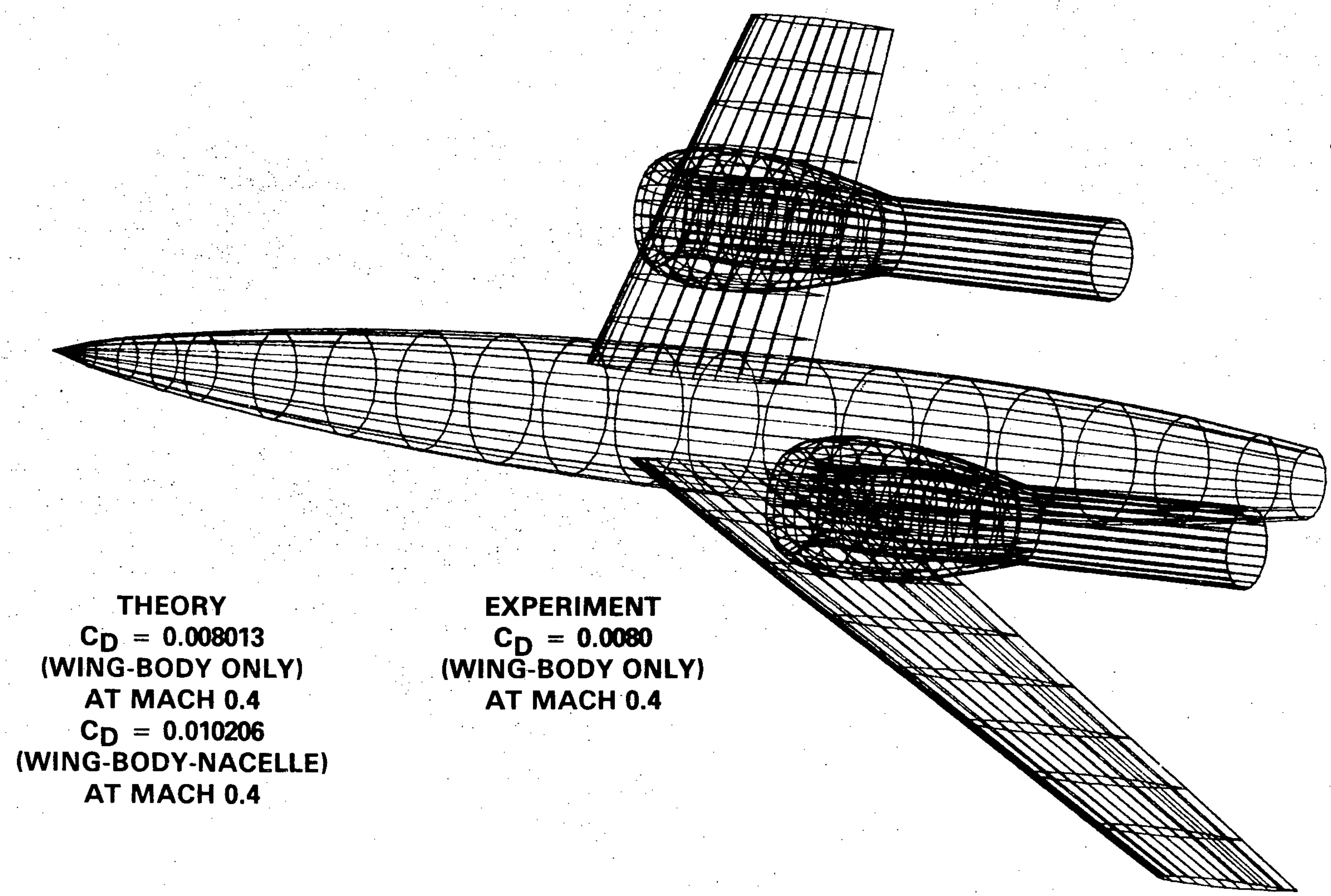
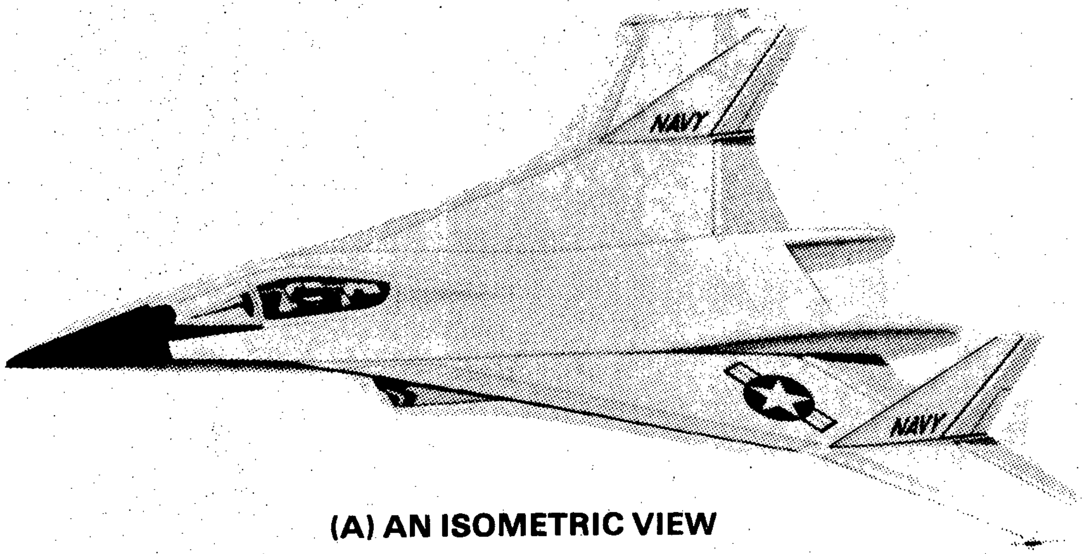
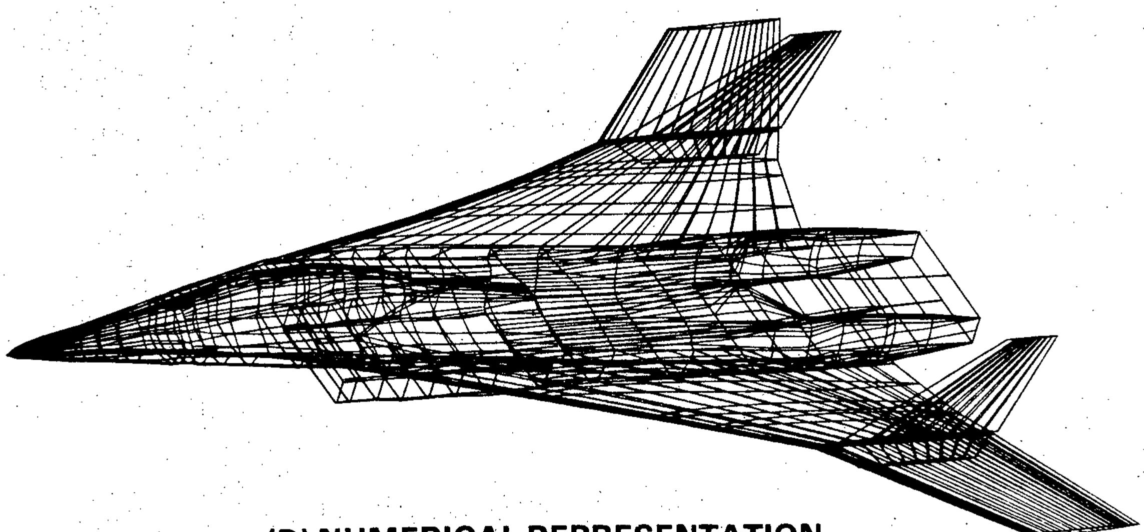


Figure 5 - Drag Prediction of a Wing-Body-Nacelle Configuration at Subsonic Flight



(A) AN ISOMETRIC VIEW



(B) NUMERICAL REPRESENTATION

Figure 6 - Drag Prediction of a V/STOL Configuration at Supersonic Flight

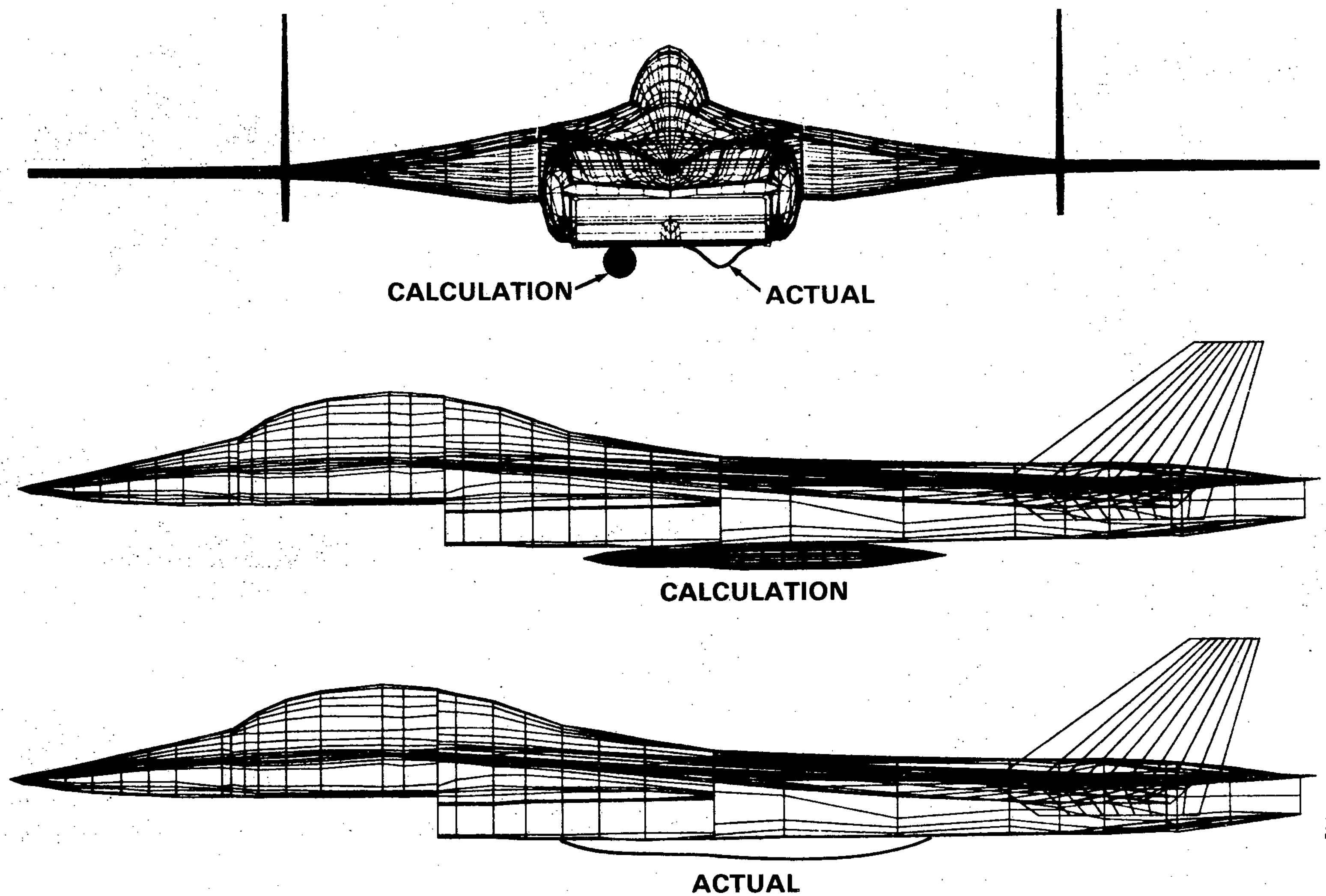


Figure 7 - Store Drag Calculation Method

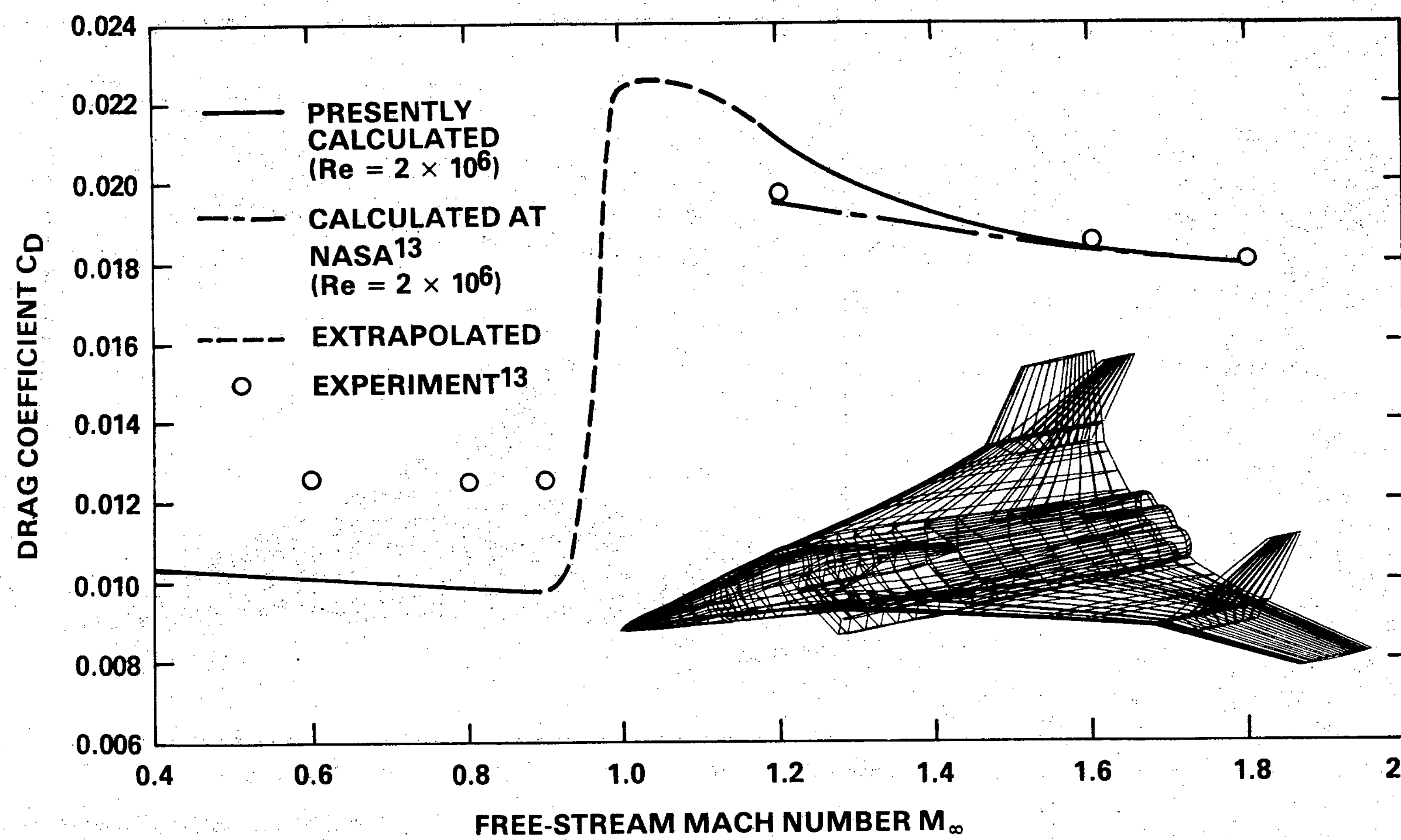


Figure 8 - Drag Coefficients of NASA Supercruiser 4 at Zero Lift

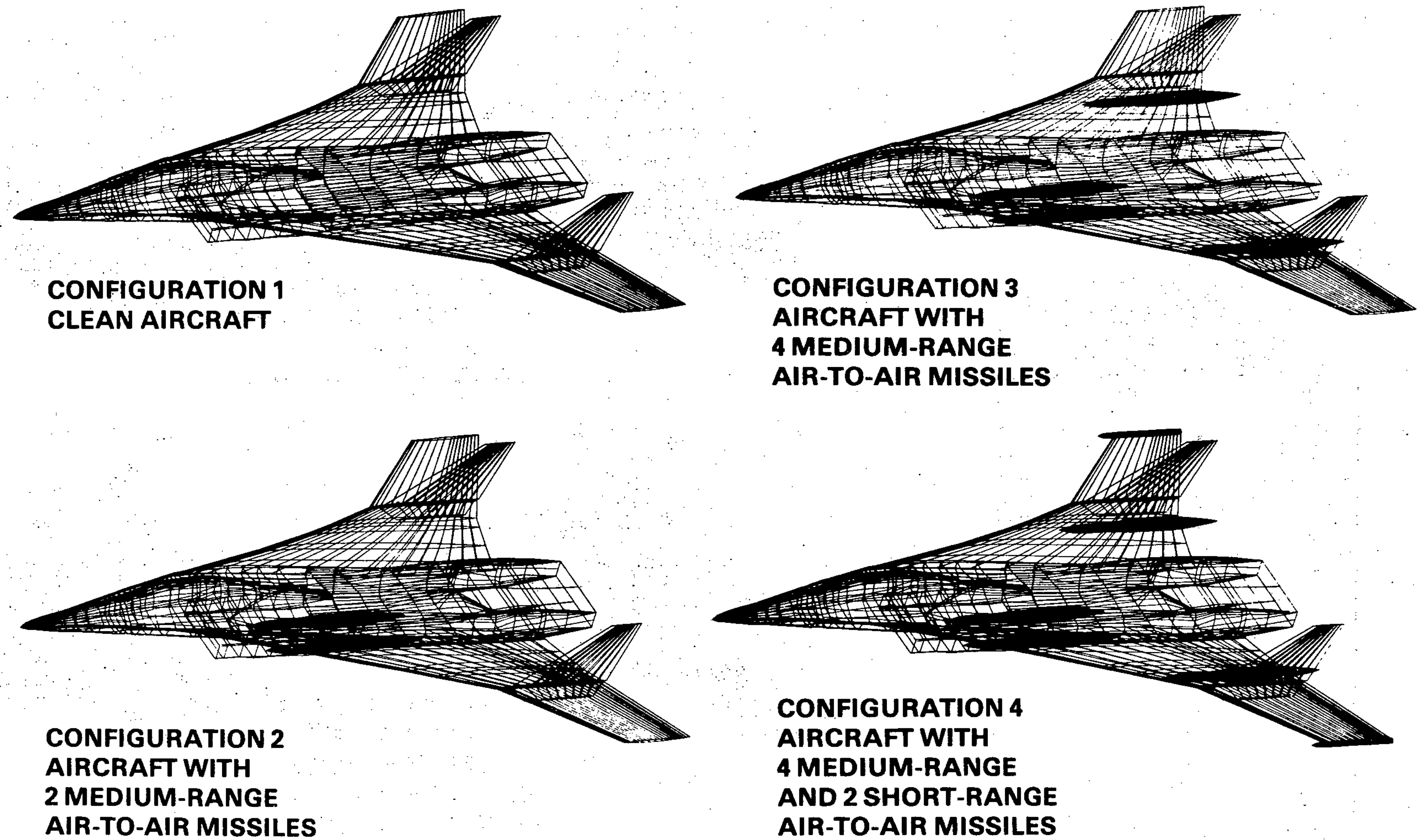


Figure 9 - Orthographic Views of a Supersonic V/STOL Configuration with Various Store Arrangements

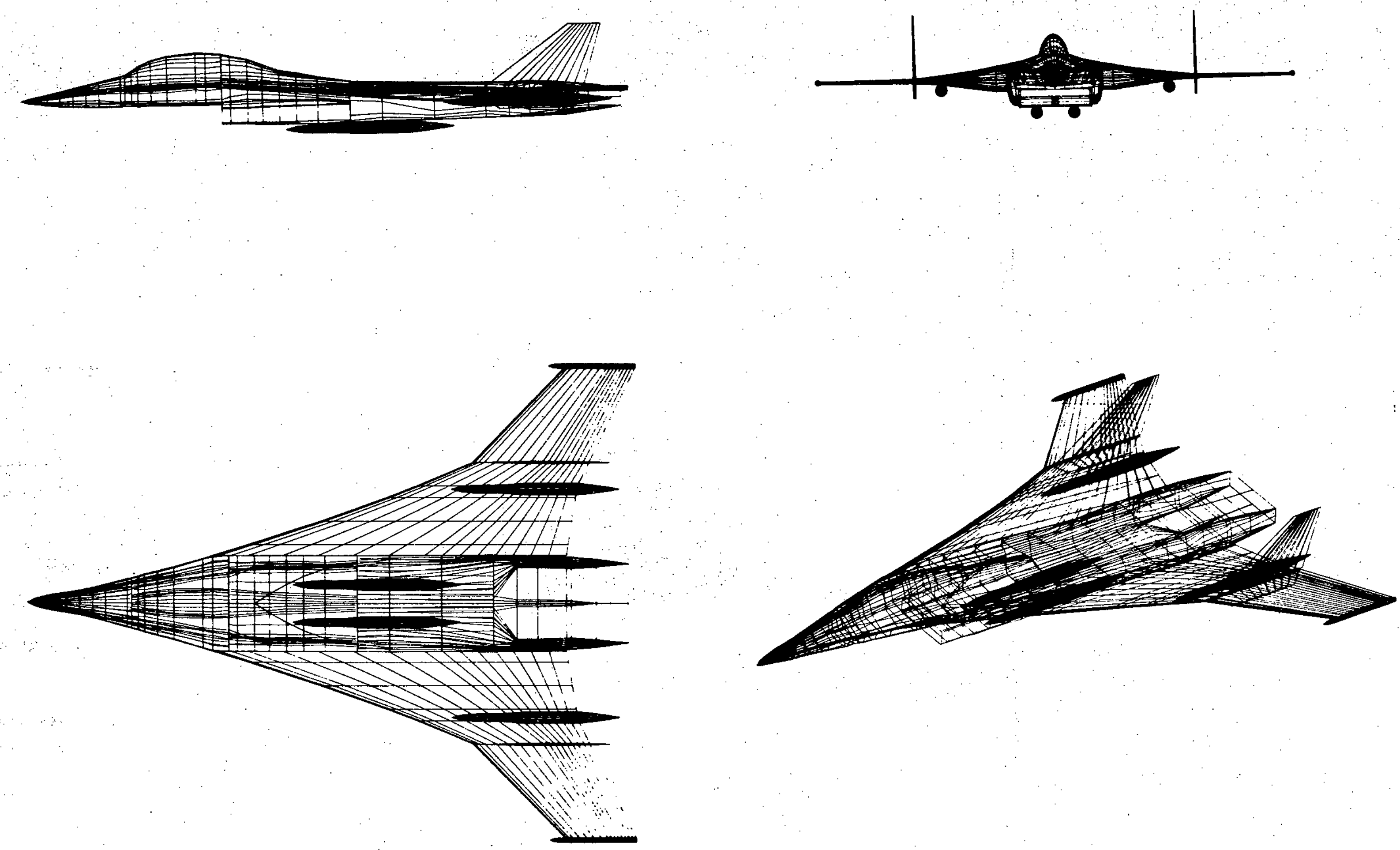


Figure 10 - Four Views of a Supersonic V/STOL Configuration with Four Medium-Range and Two Short-Range Air-to-Air Missiles

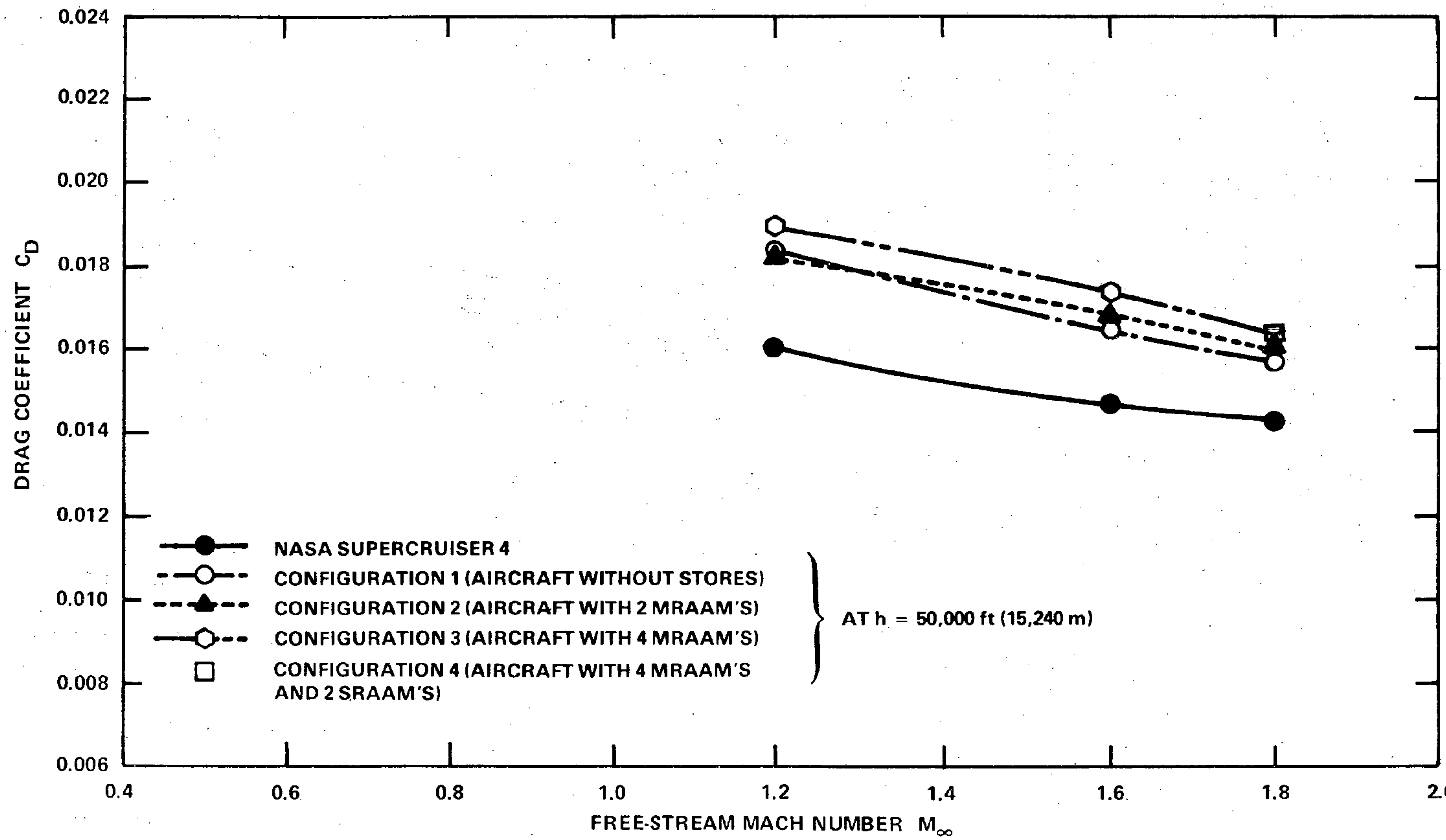


Figure 11 - Drag Coefficients of a Supersonic V/STOL Configuration with Typical Store Arrangements

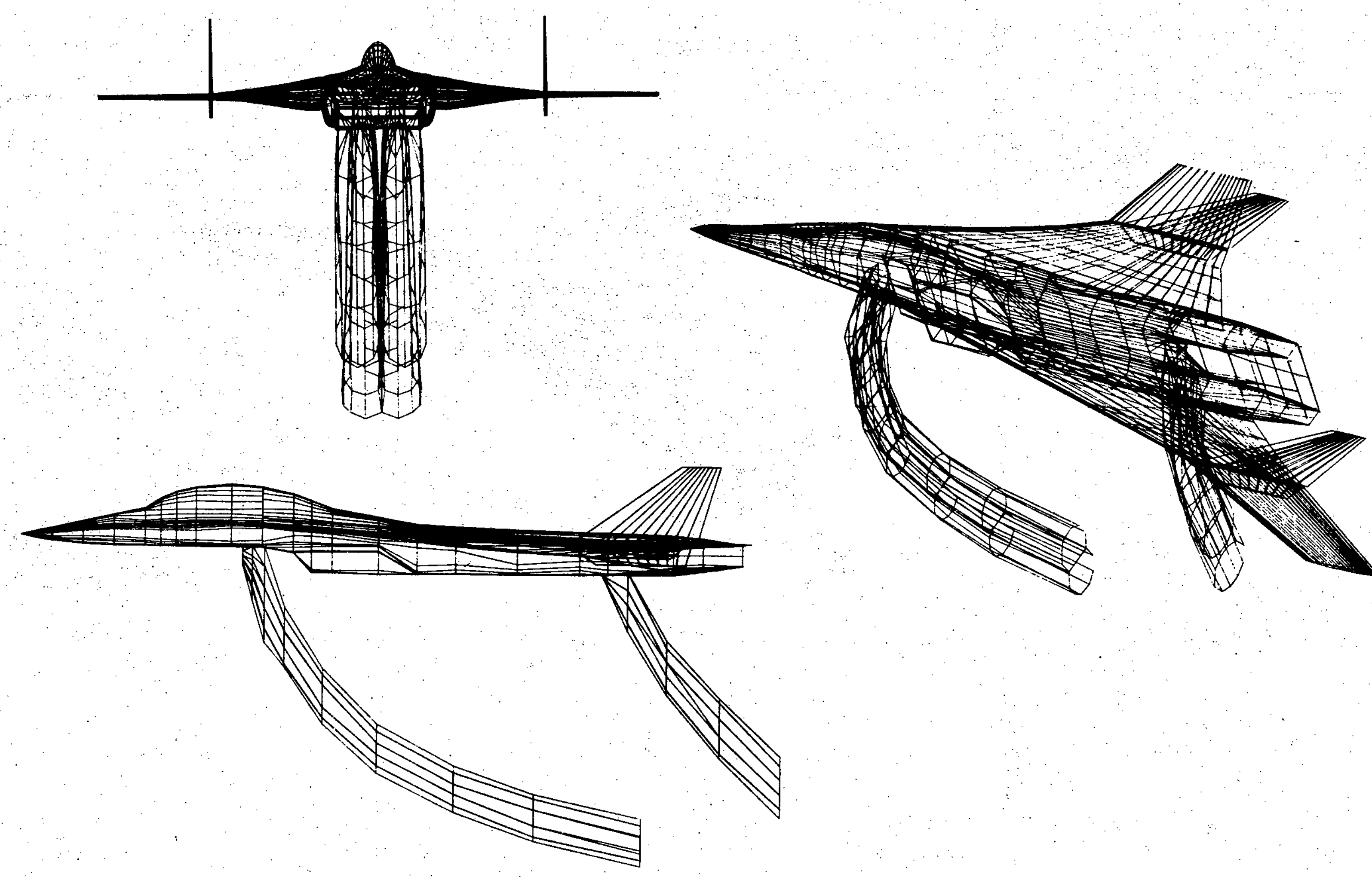


Figure 12 - A Supersonic V/STOL Configuration in Transition Mode

BIOGRAPHICAL SKETCH OF THE AUTHOR

Tsze C. Tai received his M.S. degree in mechanical engineering from Clemson University in 1965 and his Ph.D. degree in aerospace engineering from Virginia Polytechnic Institute in 1968. Upon graduation he joined the Naval Ship Research and Development Center (presently the David W. Taylor Naval Ship R&D Center). Dr. Tai has performed research in the fields of transonic aerodynamics, ejector flow, hypersonic heat transfer, and nonequilibrium flow. He is a member of the AIAA and Sigma Xi and has served as panel member on the Heat Transfer Panel and the Air Inlets and Diffusers Panel of the Navy Aeroballistics Committee.

NUMERICAL SOLUTION OF THE SUPERSONIC AND HYPERSONIC VISCOUS FLOW AROUND THIN DELTA WINGS

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Nomenclature

a	Local speed of sound
c	Adjustable constant for maximum time step
Def	Deformation tensor
e	Total specific internal energy
F, G, H	Vector fluxes
i, j	Indexes of the grid system
$\bar{\bar{I}}$	Unit identity matrix
M	Mach number
p	Static pressure
Pr	Prandtl number
$\dot{q}$	Rate of heat transfer
r	Radial coordinate
Re	Reynolds number
t	Time
T	Static temperature
U	Vector of the dependent variables
u, v, w	Velocity components in Cartesian frame
x_m, x_i	Tensor form of Cartesian coordinates
x, y, z	Cartesian coordinates
α	Angle of attack
γ	Ratio of specific heats (1.40 for air)
γ_c	Cross-flow vector angle with positive η axis
Δ	Incremental value
ξ, η, ζ	Transformed coordinate system
$\xi_{x_m}, \eta_{x_m}, \eta_{x_m}$	Coordinate transformation derivatives
θ	Conical angle
λ	Second coefficient of viscosity
μ	Absolute viscosity
ρ	Density
Σ	Summation notation
$\bar{\bar{\tau}}$	Stress tensor
$\bar{\bar{\tau}}'$	Viscous stress tensor $\bar{\bar{\tau}}' = \bar{\bar{\tau}} + p\bar{\bar{I}}$
ϕ	Circumferential angle
ϕ	Dissipation function
$\bar{\chi}$	Hypersonic similarity parameter

Subscripts

i	Tensor coordinate indexes
i, j	Indexes of the grid system
m	Coordinates indexes 1, 2, 3
max	Maximum value
o	Stagnation condition
w	Surface or wall conditions
DF	Diffusion part of linearlized Navier-Stokes Equations

Subscripts

INV	Inviscid part of linearized Navier-Stokes Equations
MXD	Mixed-derivative part of linearized Navier-Stokes Equations
∞	Freestream condition

Superscripts

$-$	Denotes vector
$=$	Denotes tensor
n	Denotes time step level

NUMERICAL SOLUTION OF THE SUPERSONIC AND
HYPERSONIC VISCOUS FLOW AROUND THIN DELTA WINGS

ABSTRACT

Numerical solutions have been obtained for the supersonic and hypersonic, viscous flow fields around thin planar delta wings. These solutions were obtained by solving the unsteady Navier-Stokes equations subject to a conical approximation. The integration technique used was MacCormack's explicit finite-difference scheme. Solutions were obtained for the upper, lower, and total flow fields around delta wings with supersonic leading edges. These solutions span a Mach number range of 2.94 to 10.16, a local Reynolds number range of 3.345×10^5 to 5.0×10^6, and various angles of attack from -15° to +15°. The numerical results compare quite favorably with both supersonic and hypersonic experimental flow field data. This investigation demonstrated the feasibility of applying a conical approximation to the Navier-Stokes equations in order to calculate the flow around thin delta wings.

Introduction

Aerodynamicists have been investigating the supersonic and hypersonic flow fields around delta wings for a number of years[1,2,3]. Recent development of the Space Shuttle has focused renewed interest and increased research activity in this area. Previous efforts at studying this problem were directed towards analyzing the lower surface or compression side of the flow field. However, from recent experiments, it was seen that the dominant features of the flow exist in the upper surface of leeside flow field. At angle of attack, an embedded shock is formed on the expansion side of the delta wing. The interaction of this embedded shock wave with the attached boundary layer can create vortices in the boundary at low angles of attack and flow separation at large angles of attack[4]. Both of these features increase the temperature and pressure gradients on the upper surface of the wing. Previous analytical and numerical studies have addressed only the inviscid supersonic and hypersonic flow around delta wings[5-9]. No satisfactory solution has been found which can calculate the induced vortex phenomena within the boundary layer and the shock-induced separation on the upper surface[3].

In this study, a unique technique is used to solve both the inviscid and viscous flow around thin delta wings. This technique involves solving the Navier-Stokes equations subject to a conical approximation[10]. The delta wings, used in this investigation, have supersonic leading edges with laminar boundary layer flows. Solutions are obtained for both the upper and lower surface flow fields at various angles of attack, Mach numbers, and Reynolds numbers. These numerical results show for a first time the shock-induced vortex development within the boundary layer and also provide good agreement with the experimental data of Bannink et al[11], Cross[12], and Spurlin[13]. These solutions provide a unique insight and and improved understanding into the dynamics of this complex flow field phenomena.

Governing Equations

The unsteady, compressible, three-dimensional Navier-Stokes equations can be written in conservative form as follows[14].

$$\frac{\partial \rho}{\partial t} + \nabla \cdot (\rho \bar{u}) = 0 \qquad (1a)$$

$$\frac{\partial (\rho \bar{u})}{\partial t} + \nabla \cdot (\rho \bar{u}\bar{u} - \bar{\bar{\tau}}) = 0 \qquad (1b)$$

$$\frac{\partial (\rho e)}{\partial t} + \nabla \cdot (\rho e \bar{u} - \bar{u} \cdot \bar{\bar{\tau}} + \bar{\dot{q}}) \qquad (1c)$$

where the heat flux is defined as $\bar{\dot{q}} = -k \nabla T$, the viscous dissipation function Φ is given as $\bar{\Phi} = \bar{u} \cdot \bar{\bar{\tau}}$ and the stress tensor is

$$\bar{\bar{\tau}} = -p\bar{\bar{I}} + \lambda (\nabla \cdot \bar{u})\bar{\bar{I}} + \mu \, \mathrm{Def} \, \bar{u} \qquad (2)$$

The equation of state, Sutherland's viscosity formula, and a fixed Prandtl number (0.72) provide a closed system of governing equations.

A generalized body-orientated coordinate transformation is used to map the physical plane (x, y, z) into the transformed plane (ζ, η, ξ). In this study, the generalized coordinate transformation is introduced as

$$\begin{aligned} \zeta &= \zeta(r) \\ \eta &= \eta\left(\frac{y}{x},\frac{z}{x}\right) \\ \xi &= \xi\left(\frac{y}{x},\frac{z}{x}\right) \end{aligned} \tag{3}$$

where

$$r = \overline{x^2 + y^2 + z^2}$$

The governing equations to be solved in this transformed domain are written as follows[15]:

$$\frac{\partial U}{\partial t} + \sum_m \zeta_{x_m} \frac{\partial F_m}{\partial \zeta} + \sum_m \eta_{x_m} \frac{\partial G_m}{\partial \eta} + \sum_m \xi_{x_m} \frac{\partial H_m}{\partial \xi} = 0 \tag{4}$$

where = 1, 2, 3; the ζ_{x_m}, η_{x_m}, and ξ_{x_m} denote the first-order partial derivatives of the transformed independent variables (ζ, η, ξ) with respect to the Cartesian coordinates (x, y, z). The vector fluxes U, F_m, G_m, and H_m are defined as

$$U = \begin{vmatrix} \rho \\ \rho u \\ \rho v \\ \rho w \\ \rho e \end{vmatrix} \tag{4a}$$

$$F_1 = G_1 = H_1 = \begin{vmatrix} \rho \\ \rho uu - \tau_{xx} \\ \rho uv - \tau_{xy} \\ \rho uw - \tau_{xz} \\ \rho eu - \dot{q}_x - \phi_x \end{vmatrix} \tag{4b}$$

$$F_2 = G_2 = H_2 = \begin{vmatrix} \rho \\ \rho vu - \tau_{xy} \\ \rho vv - \tau_{yy} \\ \rho vw - \tau_{yz} \\ \rho ev + \dot{q}_y - \phi_y \end{vmatrix} \tag{4c}$$

$$F_3 = G_3 = H_3 = \begin{vmatrix} \rho w \\ \rho wu - \tau_{xz} \\ \rho wv - \tau_{yz} \\ \rho ww - \tau_{zz} \\ \rho ew + \dot{q}_z - \phi_z \end{vmatrix} \tag{4d}$$

These equations are nondimensionalized by using a characteristic length, the free stream total pressure and density, and by using the maximum adiabatic free stream velocity. This results in a set of normalized governing equations which are functions of the free stream Reynolds number and a characteristic length determined by the Reynolds number.

In this investigation, a conical approximation is applied to the governing equations. Experimental data reveals a conical behavior for flows downstream of the apex region of a delta wing ($\bar{x} \leq 0\ (1.0)$) even when viscous effects are present[11,12,13,16]. When this requirement is imposed on the governing equations, it is assumed that the gradients in the radial direction are much smaller than those in the cross-flow directions (θ, ϕ).
Hence

$$\begin{aligned} \frac{\partial}{\partial r} &<< \frac{\partial}{r\partial\theta} \\ \frac{\partial}{\partial r} &<< \frac{\partial}{r\partial\phi} \end{aligned} \tag{5}$$

This is the mathematical definition of a locally conical flow. Gradients do exist in the radial direction, but they are sufficiently small to be neglected. When this requirement is imposed on the generalized transformed coordinates it becomes

$$\begin{aligned} \frac{\partial}{\partial\zeta} &<< \frac{\partial}{\partial\eta} \\ \frac{\partial}{\partial\zeta} &<< \frac{\partial}{\partial\xi} \end{aligned} \tag{5a}$$

Thus, the flow properties ρ, u, v, w, p, and T are nearly invariant along rays from the apex. However, it should be noted that the Reynolds number and the transformed derivatives are functions of ζ and thus a characteristic length is included in the computation.

By applying this conical approximation to the conservative form of the governing equations, the resulting equation set becomes:

$$\frac{\partial U}{\partial t} + \frac{\partial}{\partial\eta} \sum_m (\eta_{x_m} G_m) + \frac{\partial}{\partial\xi} \sum_m (\xi_{x_m} H_m) + \bar{H} = 0 \tag{6}$$

where

$$\bar{H} = -\sum_m G_m \frac{\partial}{\partial\eta}\left(\frac{\partial\eta}{\partial x_m}\right) - \sum_m H_m \frac{\partial}{\partial\xi}\left(\frac{\partial\xi}{\partial x_m}\right) + \sum_m \zeta_{x_m} \frac{\partial F_m}{\partial\zeta}$$

It should be noted that although the primitive variables are constant along lines of constant ζ, the gradients of the stress terms and the coordinate transformations are not zero.

Numerical Procedure

The system of equations is solved by a two-step predictor-corrector scheme originated by MacCormack[17]. In this algorithm, forward differences are used in the predictor step to approximate

$$\frac{\partial}{\partial\eta} \sum_m (\eta_{x_m} G_m)$$

$$\frac{\partial}{\partial\xi} \sum_m (\xi_{x_m} H_m) \tag{7}$$

while backward differences are used for these terms in the corrector step.

In order to evaluate the shear stress and heat flux terms appearing in the G_m and H_m matrices, backward differences are used in the predictor step and forward differences in the corrector step. A central difference operator is used for τ_{ij} and $\dot{q}_j$ in the $\bar{H}$ matrix. This differencing procedure results in a central difference approximation with second-order accuracy for all shear stress and heat flux terms.[18]

A generalized stability criteria analysis was performed on the finite-difference form of the simplified Navier-Stokes equations. This analysis, first suggest by MacCormack and Baldwin[19], accounts for both the inviscid and viscous dominant regions. The maximum allowable time step, in tensor form, is estimated to be

$$\Delta t_{max} \leq \frac{c}{\dfrac{1}{\Delta t_{INV}} + \dfrac{1}{\Delta t_{DF}} + \dfrac{1}{\Delta t_{MXD}}} \tag{8}$$

where

$$\Delta t_{INV} = \left[\left| u_i \frac{\partial\eta}{\partial x_i} \right| \frac{1}{\Delta\eta} + \left| u_i \frac{\partial\xi}{\partial x_i} \right| \frac{1}{\Delta\xi} + a\sqrt{\frac{\partial\eta}{\partial x_i}\frac{\partial\eta}{\partial x_i}\frac{1}{(\Delta\eta)^2} + \frac{\partial\xi}{\partial x_i}\frac{\partial\xi}{\partial x_i}\frac{1}{(\Delta\xi)^2} + 2\left(\frac{\partial\eta}{\partial x_i}\right)\left(\frac{\partial\xi}{\partial x_i}\right)\frac{1}{\Delta\eta\,\Delta\xi}}\,\right]^{-1} \tag{8a}$$

$$\Delta t_{DF} = \frac{\rho Pr\ Re}{\gamma}\left[\left(\frac{\partial\xi}{\partial x_i}\right)\left(\frac{\partial\xi}{\partial x_i}\right)\frac{1}{(\Delta\xi)^2} + \left(\frac{\partial\eta}{\partial x_i}\right)\left(\frac{\partial\eta}{\partial x_i}\right)\frac{1}{(\Delta\xi)^2}\right]^{-1} \tag{8b}$$

$$\Delta t_{MXD} = \frac{12\rho\ Re\ \Delta\eta\ \Delta\xi}{7\frac{\partial\eta}{\partial x_i}\ \frac{\partial\xi}{\partial x_i} + \sqrt{\frac{\partial\eta}{\partial x_i}\ \frac{\partial\eta}{\partial x_i} + \frac{\partial\xi}{\partial x_j}\ \frac{\partial\xi}{\partial x_j}}} \tag{8c}$$

$$c \leq 1.0 \tag{8d}$$

This time step size was only calculated at the beginning of each run and was used until a converged solution was attained.

Two numerical smoothing schemes were incorporated in the numerical procedure. The fourth-order density damping terms, developed by Tannehill et al[20] was adopted with modifications. Instead of using a rigorous transformation of the second-order derivatives of density into the transformed space, the second-order derivatives were approximated by the derivatives with respect to the transformed independent variables (η, ξ). The net result is an artificial viscosity-like term of the form

$$-\alpha_m\ \Delta t\ (\Delta\xi_m)^3\ \frac{\partial}{\partial\xi_m}\ \frac{\partial^2\rho}{\mu\xi_m^{\ 2}}\ \frac{\partial}{\partial\xi_m} \qquad m = 2,3 \tag{9}$$

added to the difference equations. This damping term is only of significance in regions of large density oscillations where the truncation error is already degrading the computation. Additionally, normal stress damping, developed by McRae[10], was also used. This damping technique applies a negative multiplier to the second coefficient of molecular viscosity to smooth out the starting transients within the flow. As soon as these abnormal initial transients decay, the second coefficient of viscosity is reset to Stokes' value and the calculations are continued to a steady state solution. The magnitude of all these damping terms are set so as to be effective in the inviscid flow region but not to change the viscous region behavior or modify the effective Reynolds number, appreciably.

Coordinate System and Boundary Conditions

The body-fixed coordinate system used in this investigation is shown in Figure 1. The coordinate transformation is defined as follows

$$\zeta = x \qquad \eta = \frac{y}{x} \qquad \xi = \frac{z}{x} \tag{10}$$

where x is the characteristic length or distance from the apex of the delta wing to the measurement plane. The mesh spacing in the computational domain (η, ξ) is uniformly distributed in both coordinate directions. The viscous effects, in this computational plane, are scaled based on the local Reynolds number.

The computational domain is initialized with free stream conditions at all points except on the body surface. The boundary conditions on the wing surface are

$$\begin{aligned} &u = v = w = 0 \quad \text{at } \eta \le \eta_{\substack{\text{wing}\\\text{tip}}} \quad \text{and } \xi = 0 \\ &T = T_w \qquad\qquad \text{at } \eta \le \eta_{\substack{\text{wing}\\\text{tip}}} \quad \text{and } \quad = 0 \end{aligned} \tag{11}$$

These expressions satisfy the no-slip velocity requirements and the experimentally determined isothermal conditions. The pressure distribution on the wing is obtained by evaluating the ξ-momentum equation at the wing surface. The derivatives contained in this momentum equation are approximated by second-order, one-sided finite differences. Once the pressure is obtained, the density is determined through the equation of state. The leading edge singularity is treated as a triple-valued point with upper, side, and lower pressure and density values. These leading edge pressures are determined by the normal (η and ξ) momentum equations. The three values of density are calculated from the equation of state.

A symmetry plane exists in the computational domain at $\eta = 0$. The boundary conditions on this symmetry plane are defined as

$$\begin{aligned} &\frac{\partial u}{\partial \eta} = 0 \qquad v = 0 \qquad \frac{\partial w}{\partial \eta} = 0 \\ &\frac{\partial \rho}{\partial \eta} = 0 \qquad \frac{\partial T}{\partial \eta} = 0 \end{aligned} \tag{12}$$

The finite-difference "reflection" method is used to calculate and model the primitive variables along the centerline plane.

The free stream boundary conditions are applied on the extremities of the computational domain. The flow through these boundaries is supersonic with respect to the transformed coordinates. Under these conditions, the flow variables at the exterior grid points are set equal to the free stream conditions and are held fixed throughout the entire integration process.

Numerical Results

Several different delta wing flow field problems were solved in this investigation. These included supersonic and hypersonic flows over the leeside of a planar delta wing, supersonic flow over the compression side

of a planar delta wing, and finally supersonic and hypersonic flows around thin planar delta wings. All of these solutions are documented in Reference 25; however, in this paper only three specific cases will be discussed in detail. These including supersonic flow field cases for both the expansion and compression sides of a delta wing and a hypersonic flow field case around a thin planar delta wing.

In this analysis, two simplifying assumptions were made in modeling the flow. These were (1) to assume the flows above and below the wing surface are independent of each other and (2) to neglect some of the three-dimensional flow field effects in the immediate vicinity of the wing vertex and around the leading edge of the wing. Both of these simplifications were applied to the expansion-side-only and compression-only solutions; however, only the second one was used in solving the total flow field.

Expansion Side Flow Field Solution

In order to illustrate an expansion-side-only solution, a supersonic flow field case was selected. The free stream conditions chosen for this calculation are

$M_\infty = 2.94$ $\qquad$ $T_{0_\infty} = 544°R$

$Re_x = 2.64 \times 10^6$ $\qquad$ $P_{0_\infty} = 96$ psia

$Pr = 0.72$ $\qquad$ $T_w = 199.5°R$

where the Reynolds number is based on a root chord length of 0.173 ft. These flow field parameters correspond identically to those used by Bannink and Nebbeling[11] in their experimental investigation.

The computational domain, on which these calculations were made, consisted of a $26(\eta) \times 30(\xi)$ grid array, similar to the upper half of the grid system shown in Figure 1. For a wing sweep angle of 45.3°, the η step size was 0.063158 with 14 grid points on the wing surface. In the ξ direction, a constant, but different, step size was used for each measurement height (pitot pressure measurements by Bannink,[11]) so that experimental and numerical results could be compared without interpolation. These constant step sizes ranged from 0.02393 to 0.02775 for different calculations. The results of these calculations are shown in Figures 2-6.

In Figures 2 and 3, several spanwise pitot pressure distributions are shown for various heights above the delta wing. The numerical results of this technique are compared with the inviscid calculations by Kutler[9] and the experimental measurements by Bannink and Nebbeling[11]. For ξ=0.0718 (Fig 2), the experimental pitot pressure profile across the inboard shock is clearly affected by the interference with the wing boundary layer. At this height, the measured shock strength is slightly less than that measured at higher

values of ξ. This numerical technique accurately predicts the shock wave-boundary layer interaction, which is not accounted for in Kutler's solution. Both theoretical methods provide good agreement with experimental data, except in the vicinity of the bow shock. In this region, the current numerical model does not account for the spill over of the compression side bow shock at the leading edge. By neglecting the lower surface flow influence, a much weaker leading edge compression wave is computed due to displacement effects of the boundary layer. This very weak shock wave is clearly depicted in the density contour plot shown later in Figure 6.

Since the inviscid flow field and shock wave structure are nearly concial, the velocity components in a spherical coordinate system are used to delineate the three-dimensional flow field. The velocity vectors normal to the rays through the vertex of the delta wing are projected on the ξ-η plane as shown in Figure 4. The relative magnitude of these vectors is illustrated for all but the lowest momentum region. The direction of these vectors is given by

$$\tan \gamma_c = \frac{u\xi - w}{u\eta - v} \tag{13}$$

where γ_c is the angle measured from the positive η axis. The locus of these velocity vectors trace out "pseudo" streamlines which converge at a vortical singularity point (cross-flow stagnation point) near the origin. In the unperturbed flow region (upper and right portions of Figure 4), the conical cross-flow velocity vectors point to the origin of the coordinate system. Figure 5 illustrates some of the cross-flow conical streamlines (780 points). A linear interpolation technique (i.e. modified Euler-Cauchy method) was used to determine these steady state path lines. Figure 5 shows more clearly how most of these conical streamlines converge at the cortical singularity point, even though the numerical resolution in the boundary layer is marginal. This result is consistent with the experimental observations by Bannink and Nebbeling.

The density contour in the physical y-z plane is shown in Figure 6. This contour plot was developed by using a General Purpose Contouring Program[26] on the CDC 6600 computer. A total of 780 data points were evaluated in order to produce this figure. The internal shock wave and leading edge expansion fan are clearly depicted as highly concentrated contour lines. These contour lines indicate that the inboard shock wave, starting perpendicularly from the wing surface, extends into the central region where the expanded flow is dominant. In this region, the internal shock weakens and eventually becomes a conical sonic line. Along the wing surface, the boundary layer is very thin and the adverse pressure gradient induced by the shock wave-boundary layer interaction is weak. No flow separation occurs at the base of the internal shock.

Compression Side Flow Field Solution

In this study, only one compression side flow field case was computed. The free stream conditions chosen for this calculation are

$M_\infty = 4.0$ | $\Lambda = 50°$

$Re_x = 5.0 \times 10^6$ | $T_W = 530°R$

$Pr = 0.72$ | $\gamma = 1.4$

$\alpha = -15°$ | $\bar{\chi} = 0.028$

where T_W is equal to the free stream stagnation temperature. These flow conditions are identical to those used by Babaev[21], Voskresenskii[22], Beeman and Powers[24], and South and Klunker[23] in their inviscid analyses.

A $26(\eta) \times 45(\xi)$ array was used in this numerical calculation. This constant step size array was identical to the lower half of the grid system shown in Figure 1. The η step size was 0.059936 with 14 grid points on the wing surface. The ξ step size was 0.00568. The free stream boundary locations were positioned far enough from the wing surface and bow shock so as to not affect the numerical solution. The results of this calculation are shown in Figures 7-9.

In Figure 7, the coefficient of pressure on the lower surface of a flat delta wing is plotted for various spanwise locations. The primary area of interest, in this figure, is the subsonic cross-flow region, where a variation in surface pressure occurs. The cross-flow sonic line (as seen later in Figure 8) serves as a dividing line between the rotational and irrotational portions of the flow. Close agreement is seen between the subsonic numerical calculations and all of the analytical solutions, except Babaev's solution. In Babaev's method, an attempt is made to account for the singularity which occurs at the cross-flow sonic point (on the wing surface). The analytical surface pressure distribution should exhibit a "corner" or slope discontinuity at the cross-flow sonic point. However, Babaev's solution as well as all the other analytical solutions, show a very smooth pressure distribution at this point. Most of the other analytical techniques ignored this weak singularity.

In Figure 8, the cross-flow Mach number contours based on the magnitude of the conical cross-flow velocity components are shown. This figure defines the cross-flow subsonic and supersonic regions and the sonic line that separates these regions. The sonic point is located at the base of the sonic line in the boundary layer or on the wing surface (inviscid solution only).

In Figure 9, it can be seen that the shock wave from the leading edge to the sonic line is nealy planar. The inviscid, analytical shock angle, relative to the leading edge, is 21.3° compared with approximately 18.5° numerically. A strong density gradient is calculated in the thin boundary layer region, as seen by the highly concentrated density contour lines.

Total Flow Field Solution

For a total flow field solution, a hypersonic flow field case was selected. The free stream conditions chosen for this calculation are

$M_\infty = 10.17$	$T_{0_\infty} = 1780°$
$Re_x = 3.345 \times 10^5$	$P_{0_\infty} = 596$ psia
$Pr = 0.72$	$\gamma = 1.4$
$\overline{X} = 9.4$	$T_W = 1259°R$
$\alpha = 15°$	$\Lambda = 75°$

These are the same flow conditions used by Cross[12] in his experimental study of the expansion side flow field over a flat delta wing at hypersonic speeds.

The computational domain, as shown in Figure 1, consisted of a $26(\eta) \times 120(\xi)$ grid array with 75 rows of ξ-grid points above the wing and 44 rows below the wing. The η step size was 0.019139 with 14 grid points on the wing surface. The ξ step size was 0.00568. The free stream boundaries (upper, lower, and right side) were located far enough from the wing surface as to not effect the numerical solutions. The incremental step sized were selected so that no interpolation was required in order to compare experimental and numerical results. The results of these calculations are shown in Figures 10-16.

In Figure 10, a comparison is made between the experimental and calculated edge of the viscous region. This viscous profile is determined by evaluating the impact pressures in the ξ-direction, similar to what was done in Ref 12. It can be clearly seen that there is a progressive increase in the extent of the viscoud region as the angle of attack is increased. At low angles of attack ($\alpha \leq 5°$), the calculated profile can be approximated by $\delta \sim \eta^2$. This result is similar to the qualitative flow behavior noted by Rao and Whitehead[4] in their vapor screen studies. However, from Cross' data, the experimental profile is nearly linear. The largest differences between experimental and calculated results occur at the plane of symmetry, where three-dimensional effects (from the wing vertex) are dominant. For $\alpha = 9°$, a centerline "trough" appears in the calculated viscous profile. This trough is generated by shock-induced vortices in the viscous region. Cross' data, for $\alpha \geq 9°$, shows a large region of low impact pressure development along the centerline of the wing. Numerical results for $\alpha \geq 11°$ verify this large low impact pressure region or viscous "hump" along the symmetry plane. The center of this viscous hump is located near the projection point of the free stream velocity vector (through the wing vertex) on the ξ-η plane.

The spanwise impact pressure distribution for 15° angle of attack and various ξ=positions is shown in Figure 11 and 12. In these figures, good

agreement is seen between the theory and experiment except near the centerline of the wing at ξ=0.15904 and ξ=0.20454. This discrepancy occurs because the calculated viscous region is slightly thicker and wider than the measured value. Good correlation is seen near the bow shock and in the inviscid region above the boundary layer.

The conical cross-flow velocity vectors are shown in Figure 13. In this figure, two vortical singularities occur near the edge of the viscous region above and below the wing surface. The cross-flow separation point on the leeside of the wing is at η=0.12383 with the experimental point at η=0.13472. A vortex is formed in the boundary layer as a result of shock-induced boundary layer separation. This vortex is more clearly depicted in the conical streamline plot in Figure 14. From this streamline plot it can be seen that several secondary vortices may exist. These secondary vortices are located near the origin and very close to the wing surface on either side of the shock wave-boundary layer interaction zone. The circulations of these secondary vortices is much weaker than the primary vortex and opposite in direction. The calculation of a primary vortex in the viscous region is consistent with the experimental observations by Cross[12].

The density and temperature contours in the cross-flow physical domain are shown in Figures 15 and 16. It can be seen in Figure 15, that there is a large bow shock that spills over from the compression side into the expansion side flow field. Behind the bow shock is a large leading edge expansion fan which interacts with the internal shock. This internal shock wave is nearly normal to the wing surface and at its lowest edge, is incident upon the upper surface of the viscous region. The pressure gradient, (normal to the wing surface) in the viscous region is approximately zero on both sides of the delta wing.

In Figure 16, a large temperature gradient, normal to the wing surface, exists on both sides of the wing. This temperature gradient is particularly strong near the leading edge of the delta wing. On the leeside of the wing, the heat transfer rate decreases from a peak at the centerline, reaches a minimum above the embedded vortex and then increases to a maximum at the leading edge. This spanwise temperature gradient is similar to the experimental observations by Narayan[27].

Conclusion

In conclusion, it can be seen that this numerical technique accurately predicts most of the basic elements of the flow field. For the first time in any delta wing calculation, the shock-induced vortex development in the boundary layer and the viscous "bubble" on top of the hypersonic boundary layer can be calculated. Both the supersonic and hypersonic numerical results compare quite favorably with experimental data[11,12] as well as with several qualitative observations[4,27]. These numerical calculations show that a three-dimensional, supersonic and hypersonic, viscous flow around a thin delta wing can be accurately approximated by using a conical flow field model.

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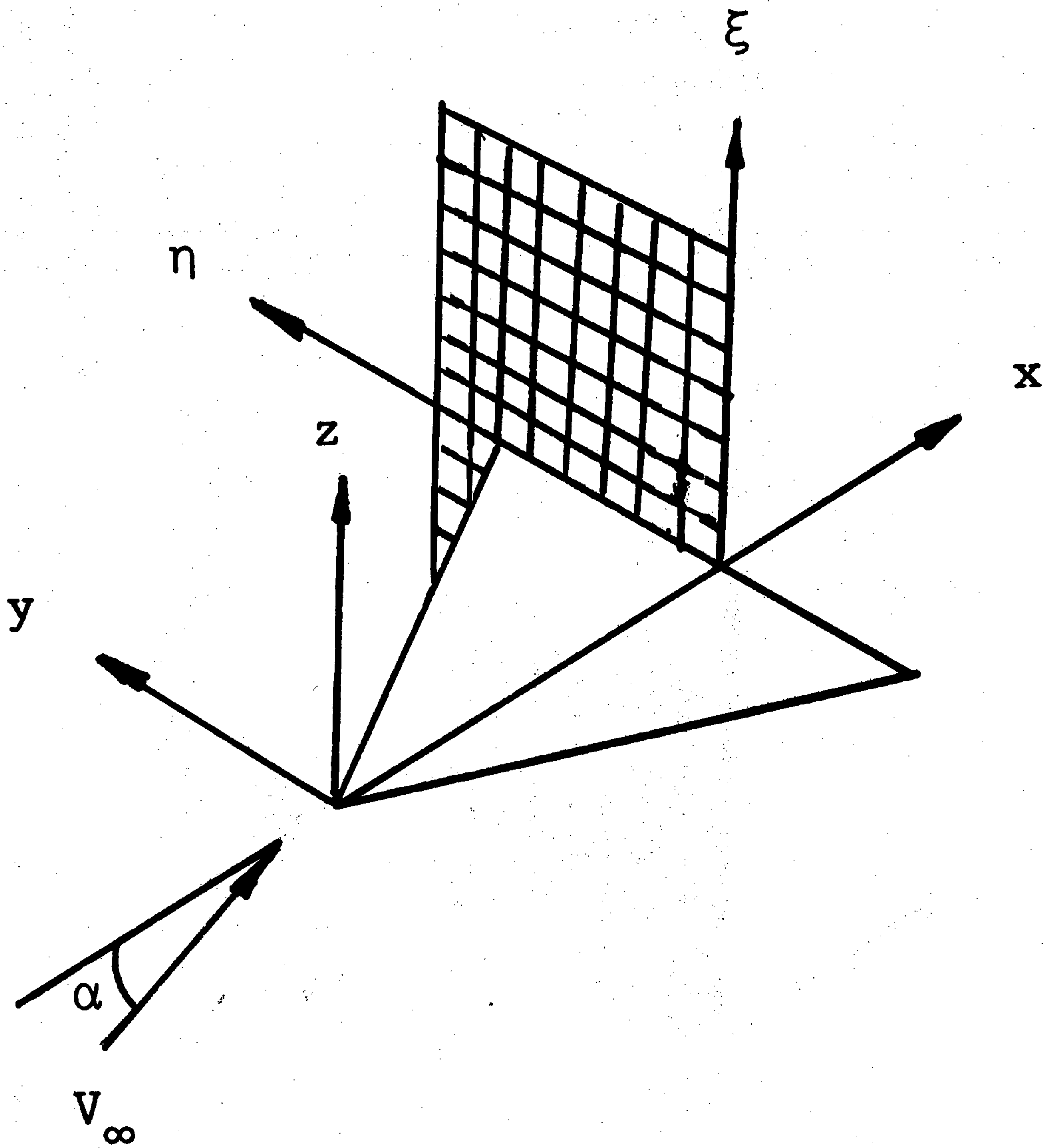

Figure 1. Coordinate System and Computational Mesh for Delta Wing.

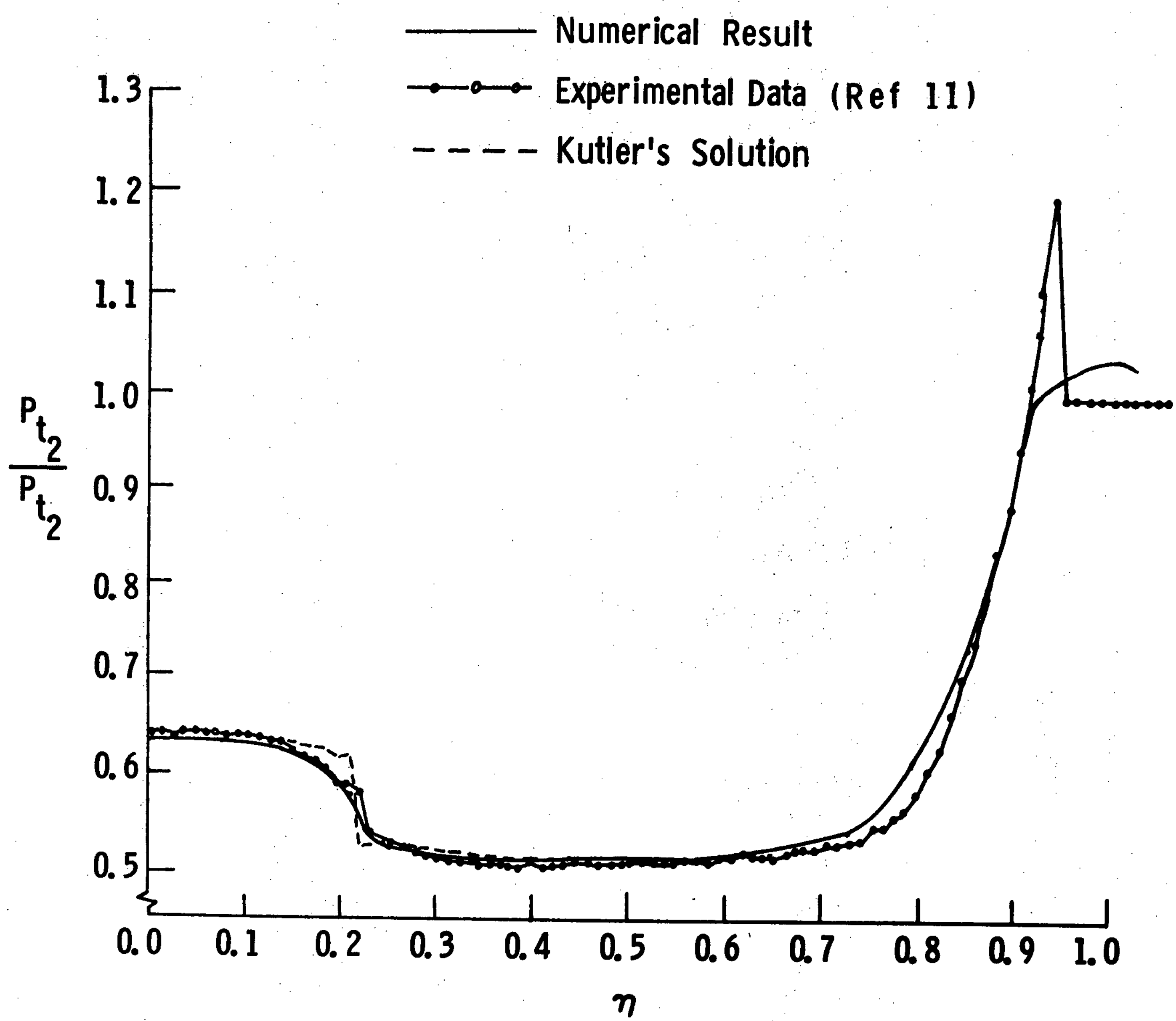

Figure 2. Pitot Pressure Distribution In Spanwise Direction for Supersonic Flow Above a Planar Delta Wing, $\xi = 0.0718$.

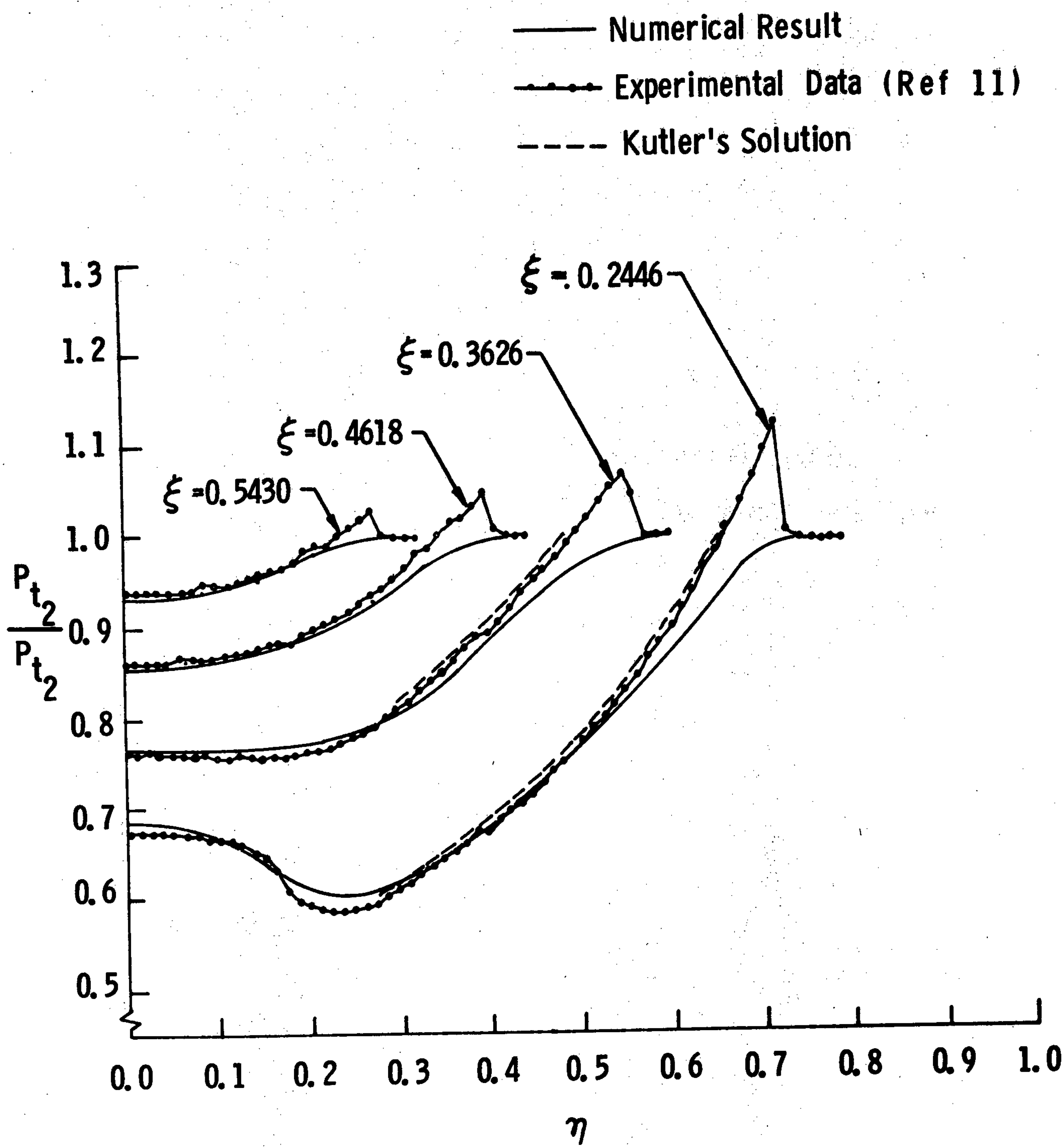

Figure 3. Pitot Pressure Distribution in Spanwise Direction for Supersonic Flow Above a Planar Delta Wing.

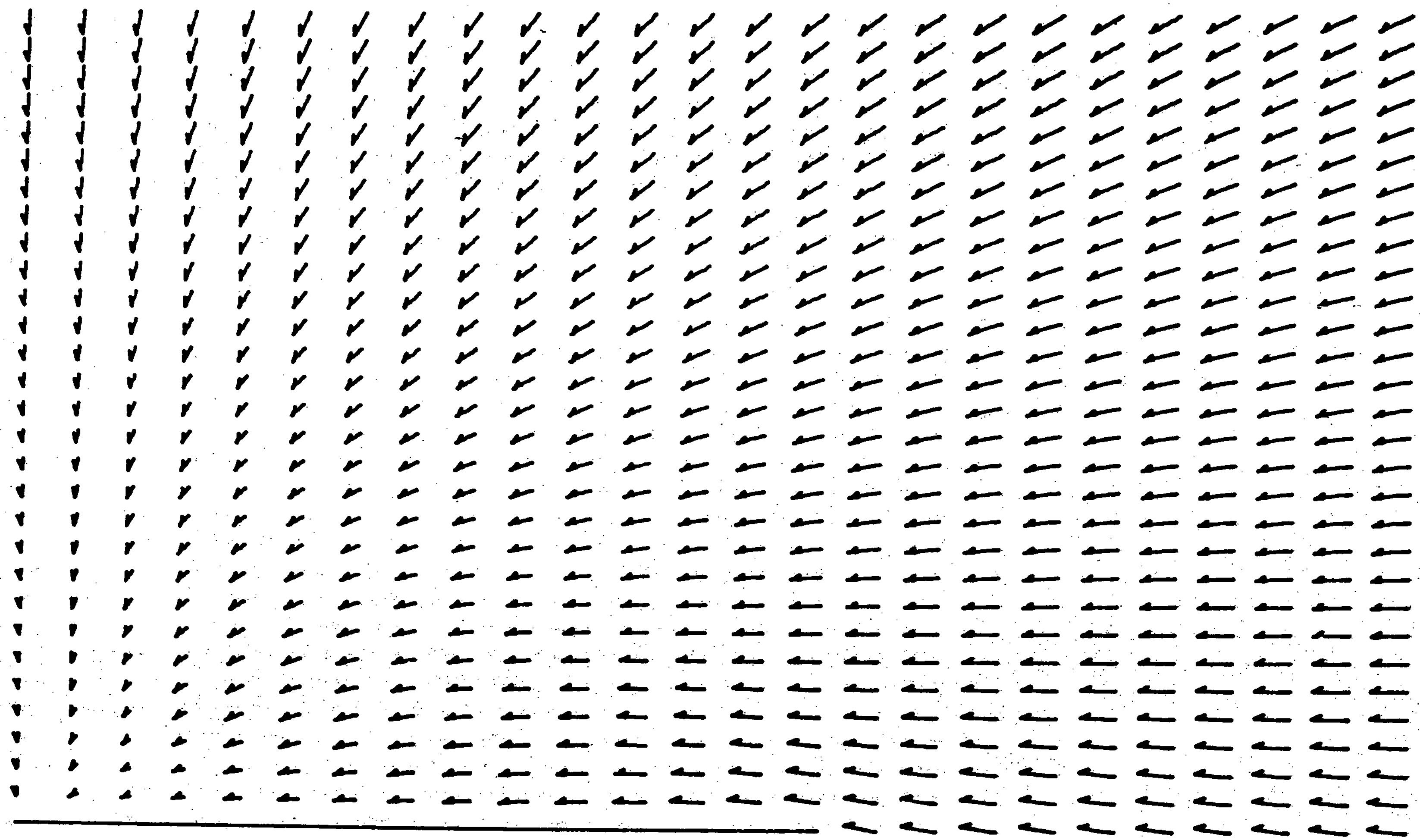

Figure 4. Cross-flow Vector Plot as seen on Spherical Surface for Supersonic Flow Above a Planar Delta Wing, $\alpha = 12^{o}$.

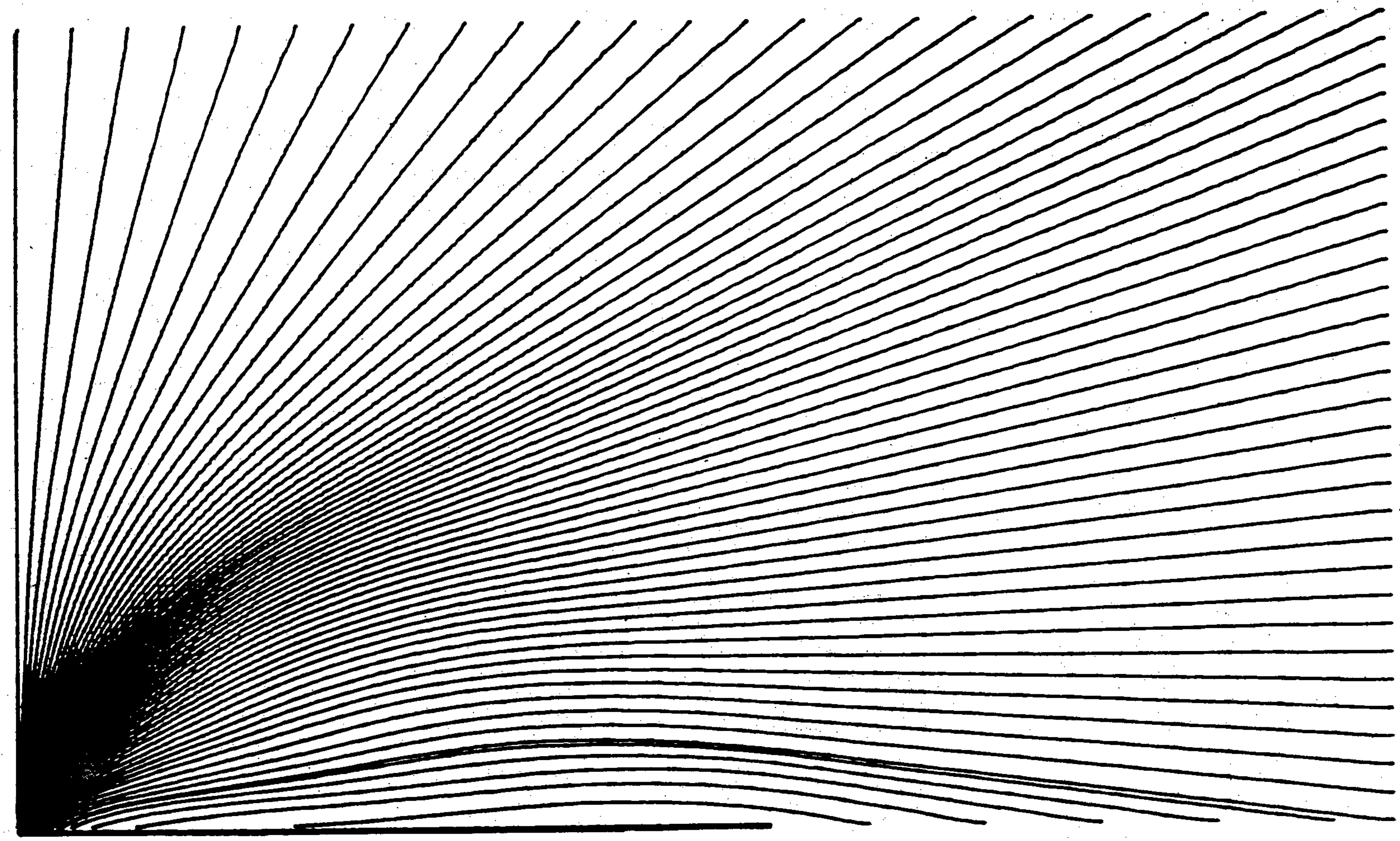

Figure 5. Cross-flow Streamline Plot as seen on Spherical Surface for Supersonic Flow Above a Planar Delta Wing, $\alpha = 12^{o}$.

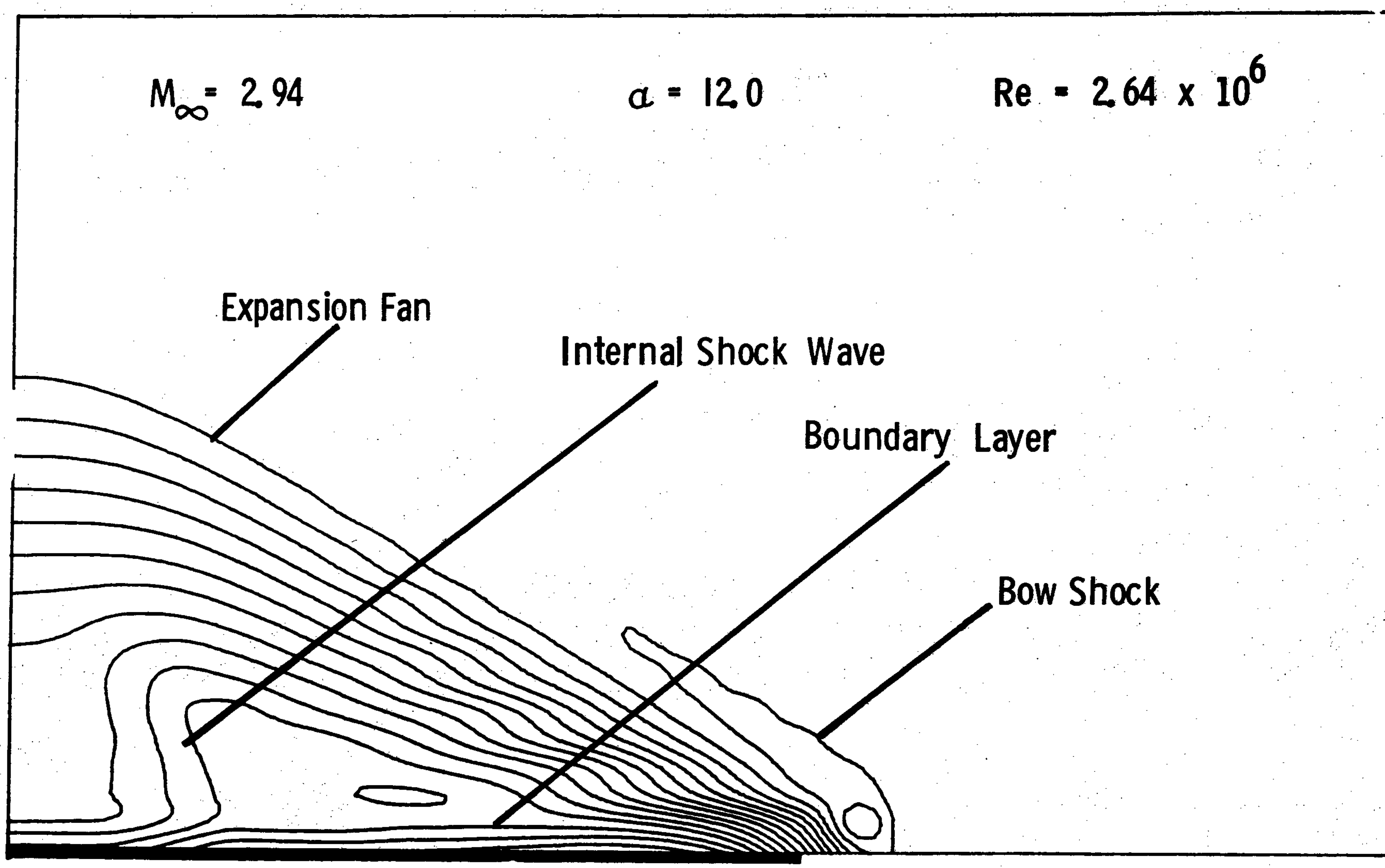

Figure 6. Cross-flow Static Density Contour for Supersonic Flow Above a Planar Delta Wing.

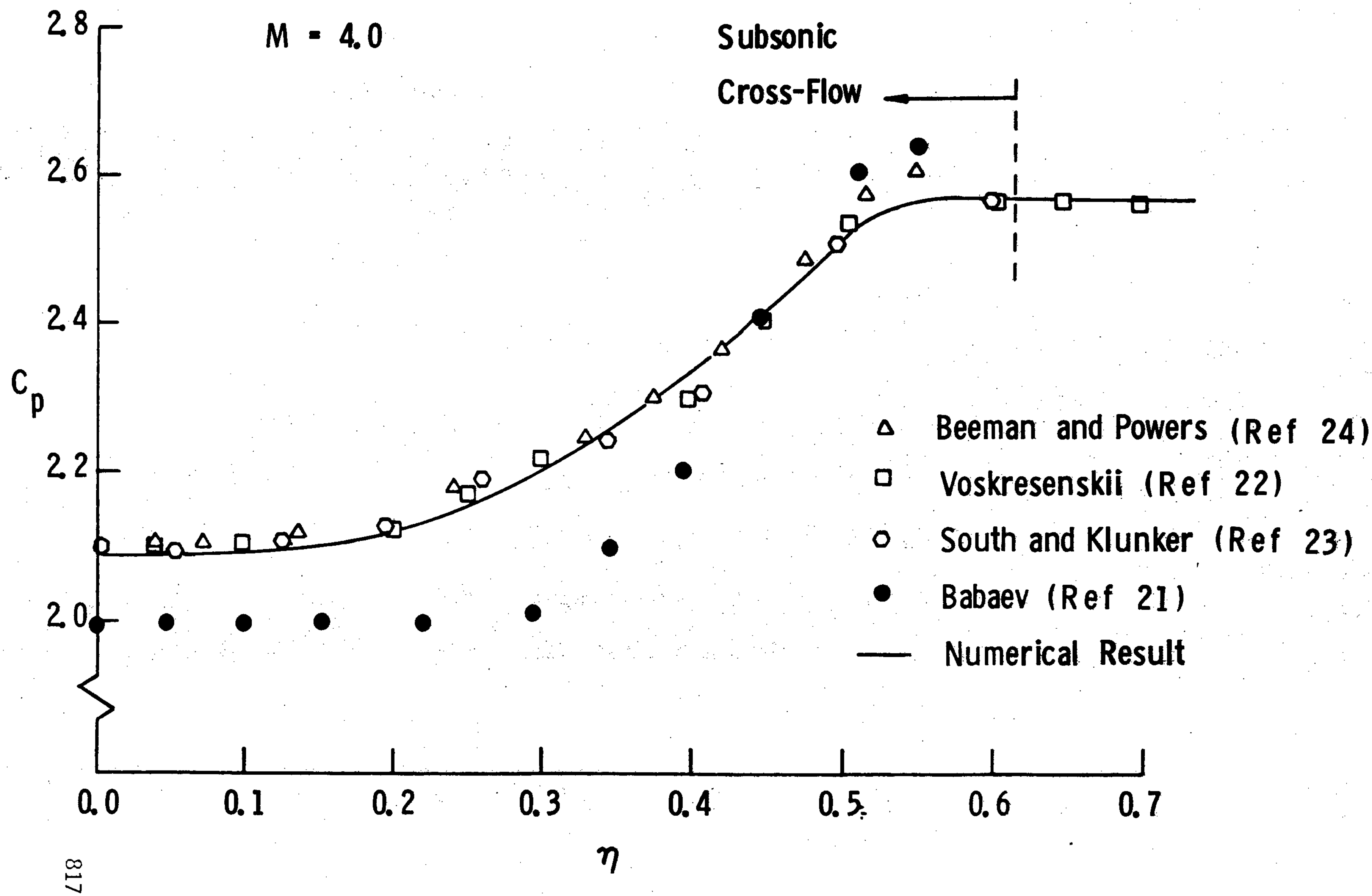

Figure 7. Spanwise Pressure Distribution on Compression Side of Planar Delta Wing at Supersonic Speeds.

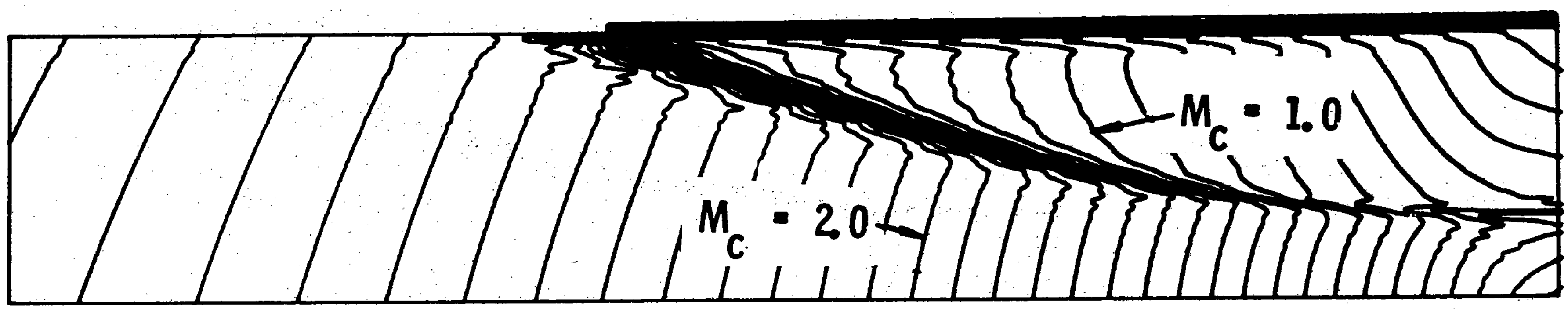

Figure 8. Cross-flow Mach Number Contour for Supersonic Flow Below a Planar Delta Wing.

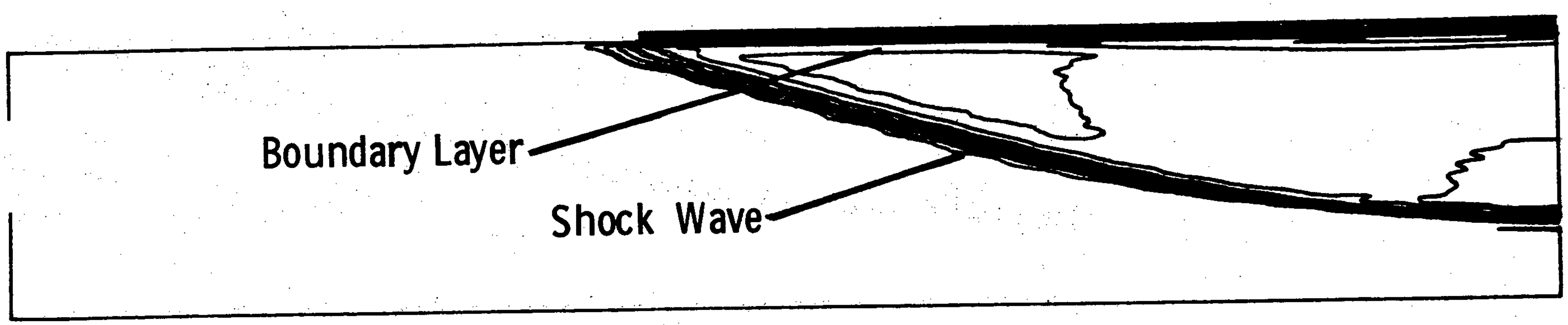

Figure 9. Cross-flow Static Density Contour for Supersonic Flow Below a Planar Delta Wing.

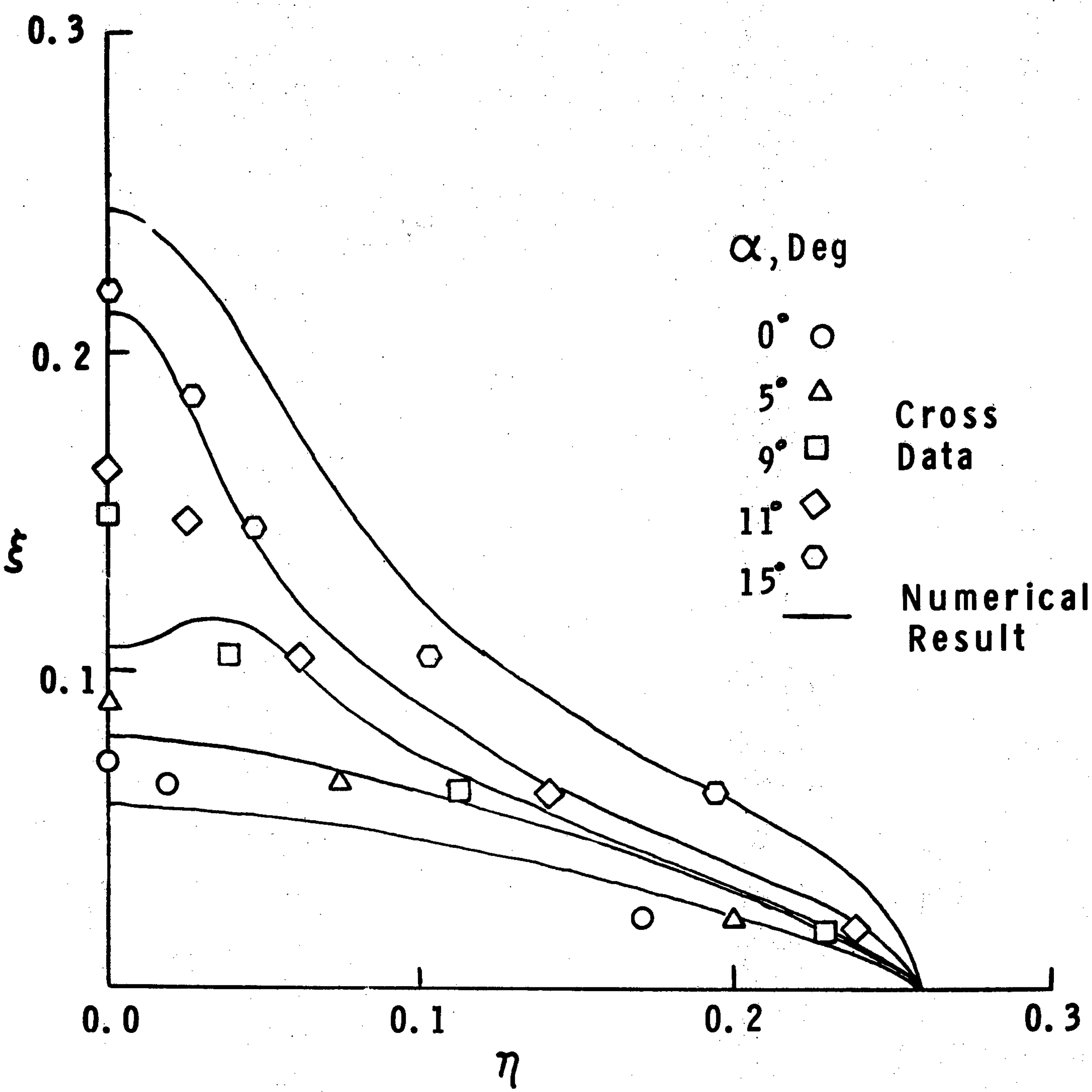

Figure 10. Hypersonic Viscous Layer Development as Determined by Impact Pressure Measurements.

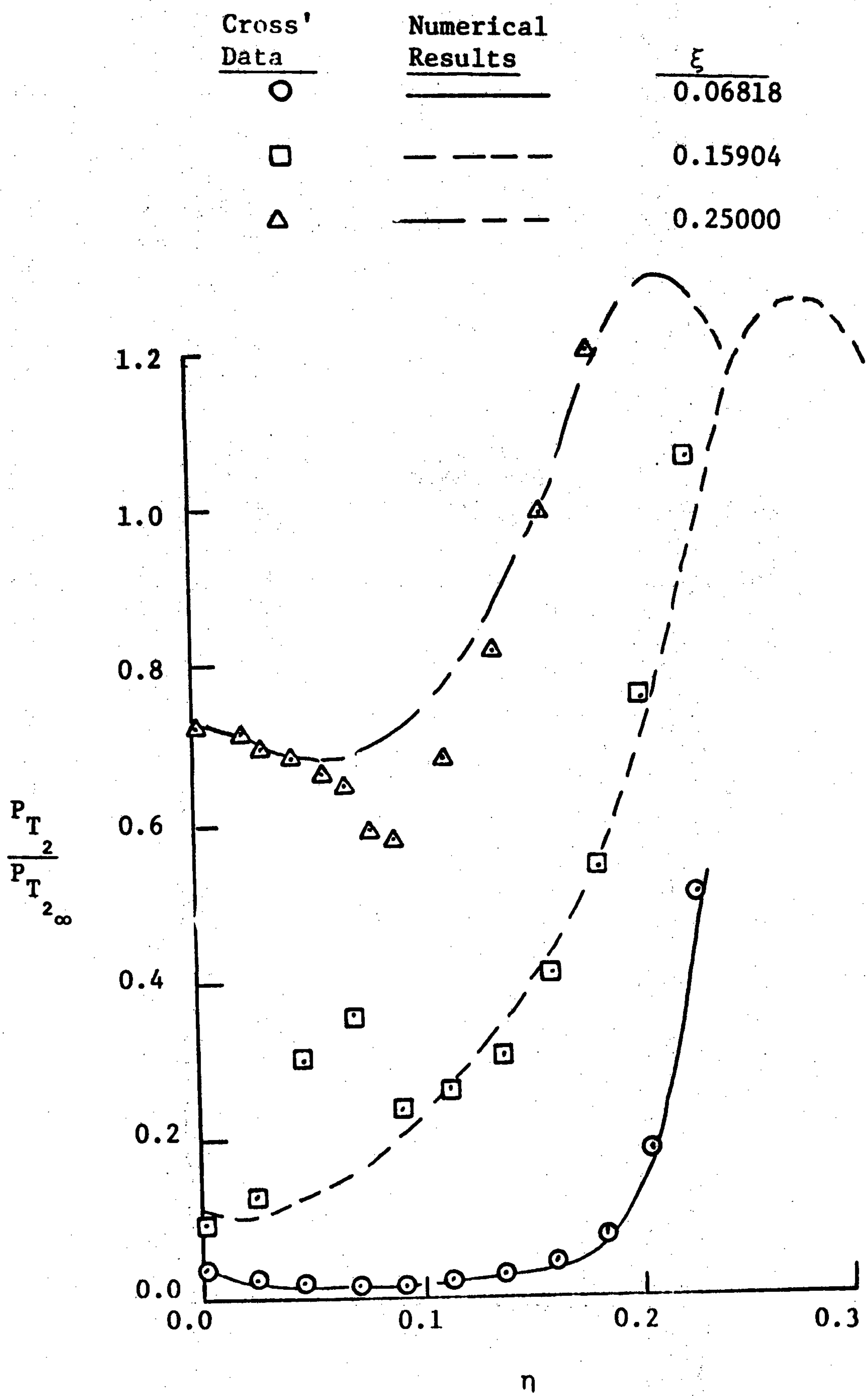

Figure 11. Impact Pressure Survey on Leeside of Delta Wing, $\alpha = 15^{\circ}$.

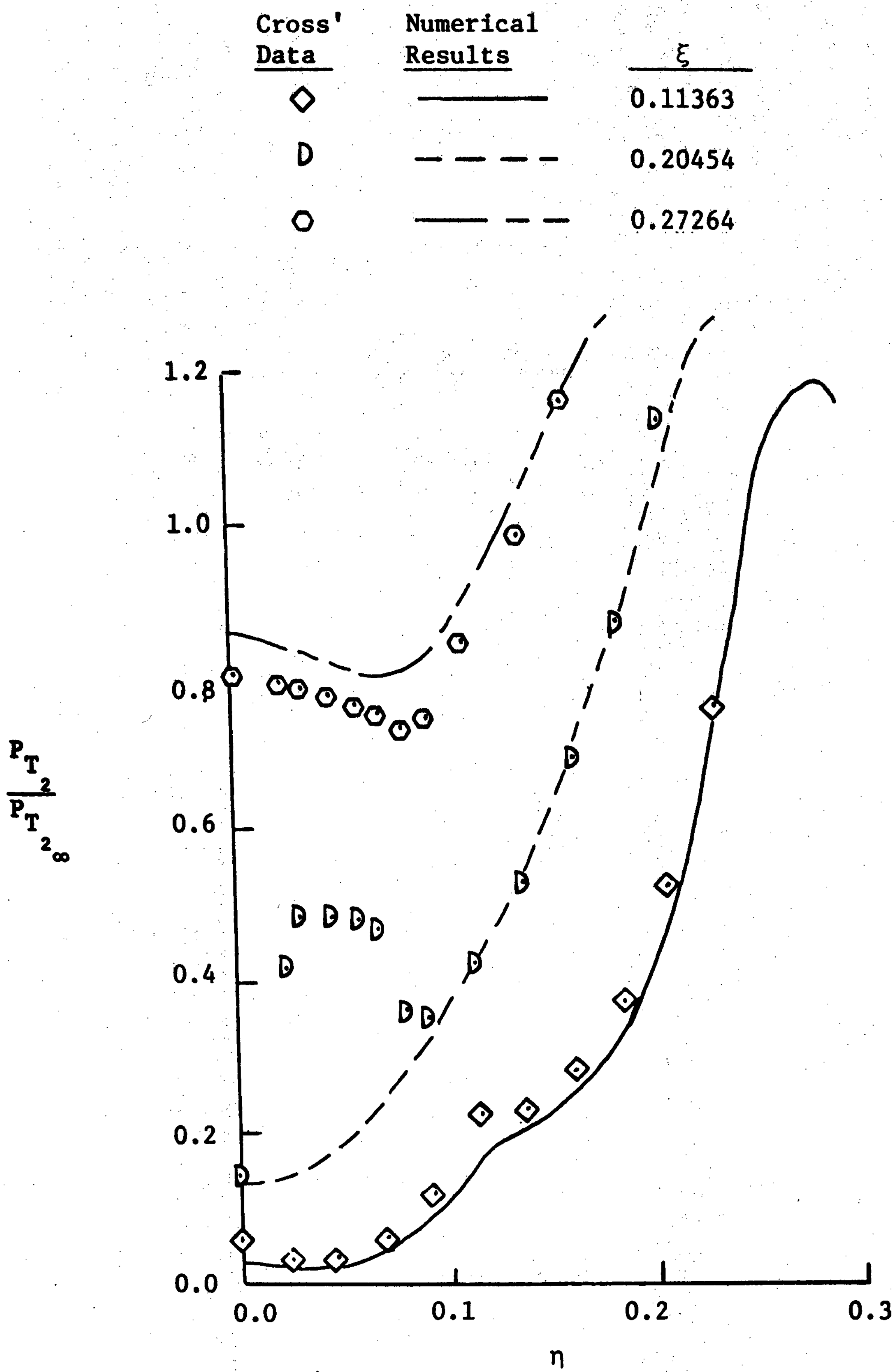

Figure 12. Impact Pressure Survey on Leeside of Delta Wing, $\alpha = 15^o$.

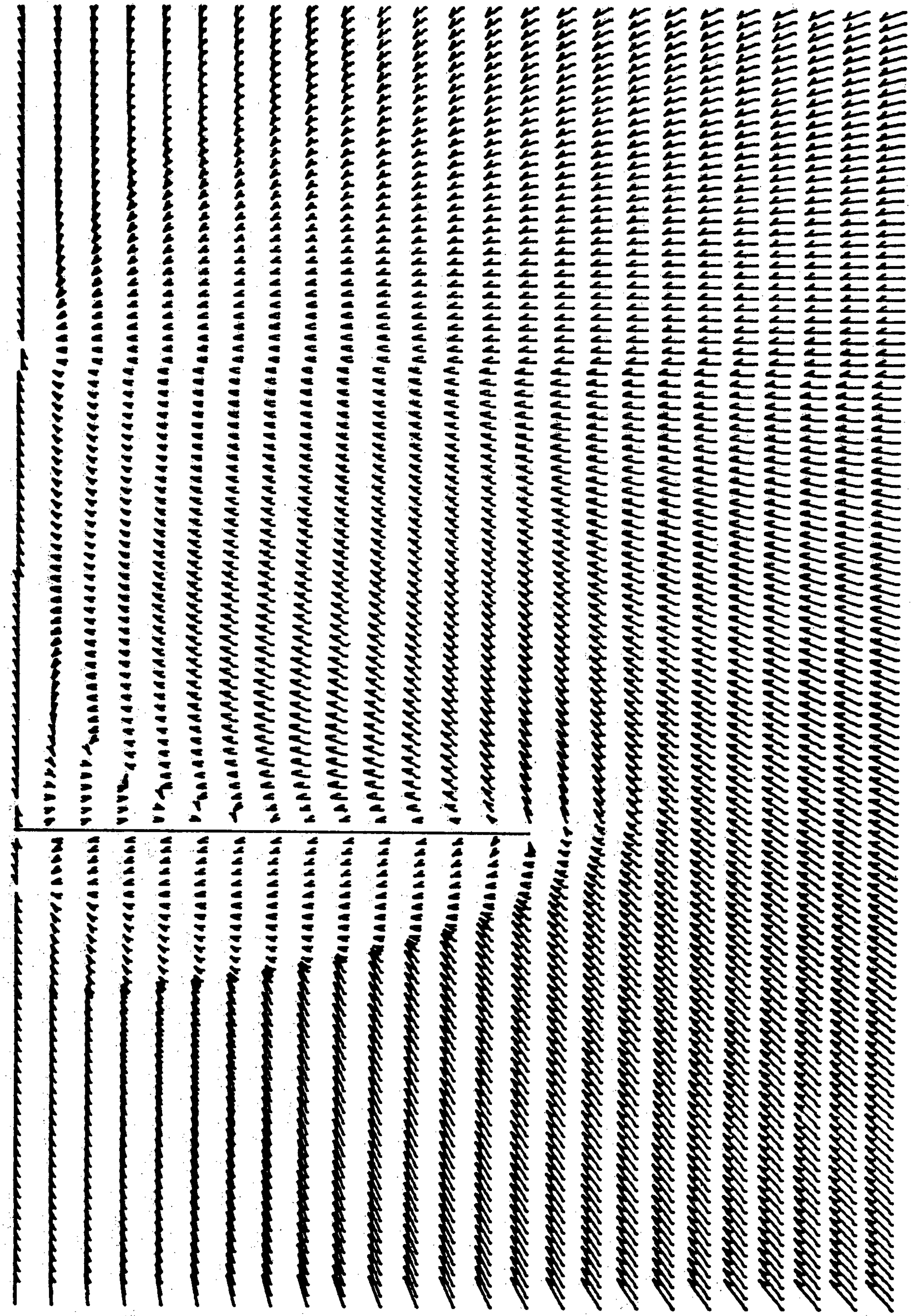

Figure 13. Cross-flow Velocity Vector Plot as seen on Spherical Surface for Hypersonic Flow Around a Planar Delta Wing, $\alpha = 15^{o}$.

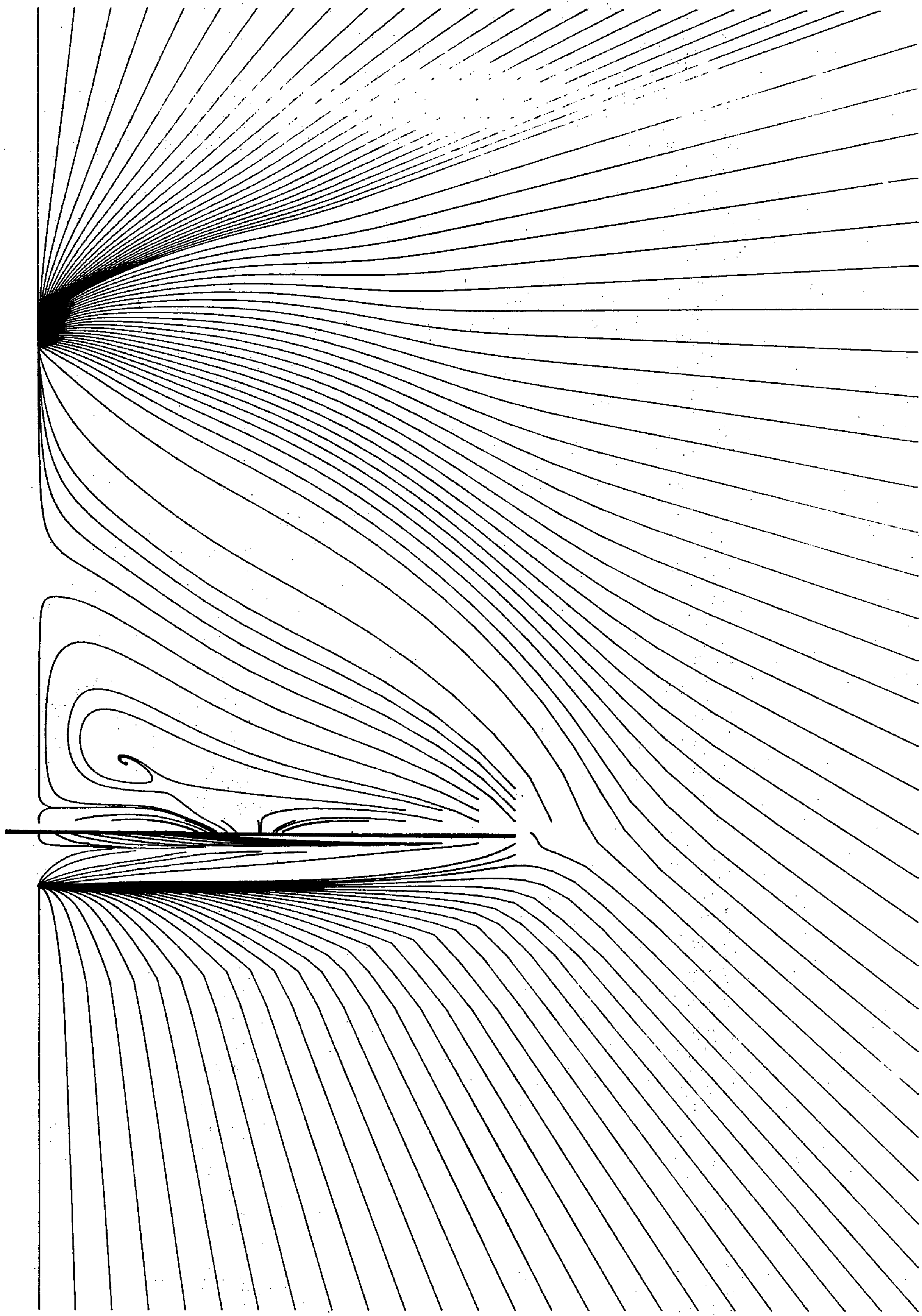

Figure 14. Cross-flow Streamline Plot as seen on Spherical Surface for Hypersonic Flow Around a Planar Delta Wing, $\alpha = 15^{o}$.

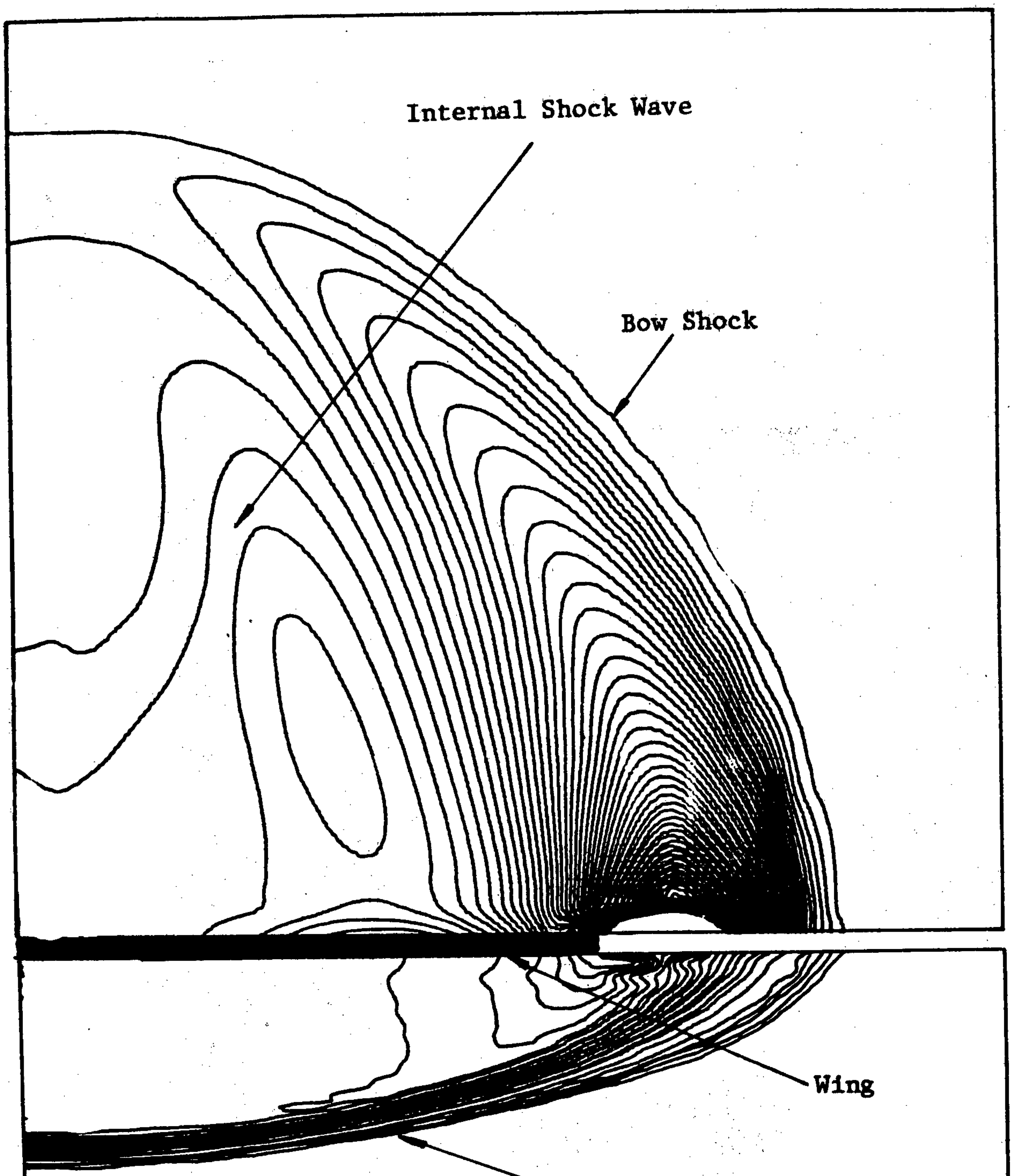

Figure 15. Cross-flow Static Pressure Contour for Hypersonic Flow Around a Thin Planar Delta Wing, $\alpha = 15^{o}$.

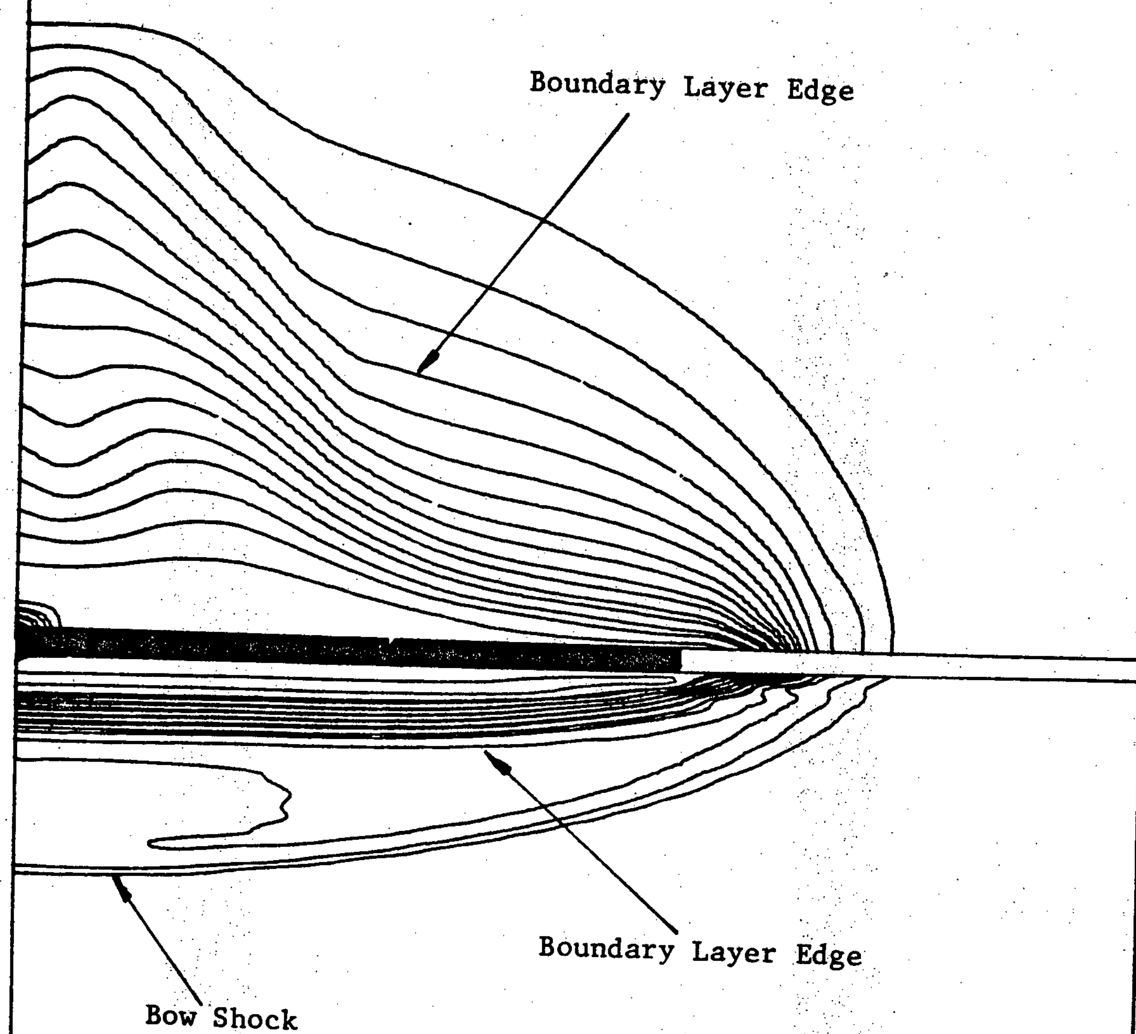

Figure 16. Cross-flow Static Temperature Contour for Hypersonic Flow Around a Thin Planar Delta Wing, $\alpha = 15^{o}$.

BIOGRAPHICAL SKETCH

Guion Stewart Bluford, Jr. was born on 22 November 1942 in Philadelphia, Pennsylvania. He graduated from Overbrook High School in 1960 and attended Penn State University where he received a Bachelor of Science degree in Aerospace Engineering and a commission in the United States Air Force in 1964. After receiving his pilot wings in 1966, he served as an F-4C pilot with the 45th Tactical Fighter Squadron, MacDill Air Force Base, Florida and with the 557th Tactical Fighter Squadron, Republic of Vietnam. In 1967, he was assigned to the 3630th Pilot Training Squadron at Sheppard Air Force Base, Texas where he served as a T-38 flight instructor and assistant flight commander. He was executive support officer to the Deputy Commander of Operations and School Secretary of the 3630th Flying Training Wing prior to entering the Air Force Institute of Technology (AFIT) in 1972. After receiving a Master of Science degree in Aerospace Engineering in March 1974, he entered the doctoral program at AFIT, where he completed the course work in December 1974. He was then assigned to the Air Force Flight Dynamics Laboratory (AFFDL/FXM) as Deputy for Advanced Concepts in the Aeromechanics Division. In April 1977, he became Branch Chief of the Aerodynamics and Airframe Branch in the Air Force Flight Dynamics Laboratory.

Dr. Hankey received his B.S. Degree in Engineering from Pennsylvania State University in 1951, M.S. Degree from the Massachusetts Institute of Technology (1953) and a PhD from Ohio State University (1962) in Aeronautical and Astronautical Engineering. Dr. Hankey entered the Air Force as a Lieutenant in 1953 and was assigned to the Transonic Wind Tunnel Unit, WADC, where he accomplished a wide range of transonic flow studies. He was appointed Chief of the Aerothermodynamic Unit of the Hypersonic Flight Section, Aircraft Laboratory, WADD. During this time, Dr. Hankey made detailed studies into hypersonic flow phenomena, especially heat transfer, and published a design handbook, DESIGN PROCEDURES FOR COMPUTING AERODYNAMIC HEATING AT HYPERSONIC SPEEDS, WADC TR 59-610. Dr. Hankey was appointed Chief of the Aerodynamics Branch, X-20 (Dyna-Soar) System Program Office in December 1960. In that position he was instrumental in developing the configuration of the X-20 and was responsible for the aerodynamic, heat transfer and performance aspects of the X-20. During 1964-1966 he was responsible for the evaluation of the Supersonic Transport Design proposals in the area of heat transfer and also taught, as an adjunct professor, Advanced Heat Transfer at the USAF Institute of Technology.

Dr. Hankey has been an aerospace research engineer in the Hypersonic Research Laboratory of the Aerospace Research Laboratories conducting theoretical studies in the optimization of hypersonic configurations. In 1967, he was invited to present a series of lectures on boundary layer theory at the AGARD von Karman Institute at Brussels. He is presently group leader in the Computational Aerodynamics Group in the Air Force Flight Dynamics Laboratory.

Dr. Hankey belongs to many honorary societies and has published over 40 technical papers in fluid mechanics. He is a member of the American Institute of Aeronautics and Astronautics, and served on the Technical Committee for Fluid Dynamics.

OPTIMIZATION OF AIRFRAME STRUCTURES: A REVIEW AND SOME RECOMMENDATIONS

BY

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Optimization of Airframe Structures: A Review and Some Recommendations

Abstract

A brief review of the structural optimization literature and a description of the present status of optimization methods is presented. The general conclusion is that the development of optimization methods and the iterative algorithms are progressing satisfactorily. However, their applications to large practical structures are slow and must be accelerated. This involves full scale service life testing and validation of optimized structures. It is also necessary to expand the present optimization methods to include some critical damage models with the object of improved survivability of optimized structures.

To illustrate the present status of optimization methods, a graphite epoxy wing structure is designed by two in-house programs and the results are presented.

Introduction

The unprecedented developments in computational capability in the last twenty-five years have fostered equally impressive developments in design optimization schemes in all disciplines of engineering. Of particular importance is the emergence of the Finite Element method for the solution of continuum mechanics problems. Interestingly, the present Finite Element method was an outgrowth of the old Moment Distribution[1] (Hardy Cross 1932) and Slope Deflection Equation Methods which were popular then for the analysis of redundant frameworks. The latter methods served the structural engineers well during the era of desk calculators. It was logical that the formal development of the Finite Element method started during the mid 50's (Argyris[2], Turner, Clough, Martin and Topp[3]) along with the development of modern digital computers. The Finite Element method provided the necessary motivation for the perfection of sparse matrix manipulation techniques, linear and nonlinear equation solvers, methods of eigenvalue analysis and numerical integration schemes. As the engineers were able to analyze more and more complex structures, it was natural to introduce optimization schemes into engineering designs. Design optimization provides not only savings in structural weight and cost but also allows interesting sensitivity studies to improve the structural integrity.

The desire for structural optimization was nothing new. The development of structural forms such as arches, domes, lattice trusses of all kinds, grillages, suspension bridges, stiffened panels, sandwich constructions, etc. is the result of engineers effort to transmit forces to various supports in an optimum manner. Besides this configuration optimization, selection of appropriate material is another important aspect of design optimization. These two aspects together constitute preliminary design. Selection of optimum sizes with various manufacturing limitations and constraints on strength and stiffness is an important step in detailed design. Interesting structural optimization schemes in this area were discussed by Maxwell (1869)[4], Cilley (1900)[5] and Mitchell (1904)[6]. The concepts of fully stressed design and simultaneous failure mode design were used to a great advantage throughout this century. However, the deficiencies of these methods were not well understood until the age of modern computers (1950's).

The development of Mathematical Programming techniques during the late 40's and early 50's provided further impetus to the application of optimization methods in engineering design. In 1960 Schmit[7] proposed coupling the Finite Element Analysis and Mathematical Programming methods for structural design optimization. In this approach structural design is treated as a problem of mathematical extremization of an objective function in an "n" dimensional design variable space constrained by such behavioral functions as stresses, displacements, frequencies etc. The search for the extremum is carried out by methods of

linear and nonlinear programming techniques. Some examples of mathematical programming methods are the gradient projection, steepest descent, feasible directions and various unconstrained minimization techniques in conjunction with the so called penalty functions to account for constraints. In the following years this approach enjoyed great popularity and was applied to a number of interesting structural design problems.

However, as the optimizers became more and more ambitious, the limitations of the mathematical programming methods proved to be too constrictive for practical applications. The elaborate mathematical transformations for determining travel directions, the sensitivity of the algorithms to step size and the initial (starting) solution are only a few of the difficulties encountered in applying mathematical programming methods to practical structural optimization problems. To offset some of these difficulties an intense effort to revive the optimality criteria approaches was started by the author and his associates in 1968. This effort paid off well as evidenced by the renewed interest in this area and subsequently resulted in very successful optimization schemes. Some of these schemes will be discussed later.

The optimality criteria approach essentially involves the derivation of necessary and sufficient conditions for an optimum with specific design requirements and associating them with a certain real or pseudo energy functional of the system. Since many of these energy functions can be readily determined from a finite element analysis, no additional (costly) transformations would be necessary for determining the travel directions. In the past ten years discrete optimality criteria were derived for a number of design conditions involving strength, static stiffness, dynamic stiffness, static stability, aeroelastic constraints, etc. The computer programs OPTSTAT[8], OPTCOMP[9], OPTIM[10], ASOP[11,12], FASTOP[13], DESAP[14], etc. use discrete optimality criteria approaches as a basis for structural optimization. All these programs were based on the finite element method. They can optimize isotropic, anisotropic and layered composite structures. These programs can handle over a thousand variables and a comparable number of degrees of freedom.

A Brief Survey

A great deal has been written on structural optimization in recent years. Extensive reviews are being published periodically on the subject, and it would be a duplication of effort to survey the entire literature. Instead this review will be limited to highlights of some recent developments and identification of various groups actively involved at present in structural optimization. This review does not attempt to include all aspects of the optimization work done by these groups, but, however, effort will be made to include one or more references that reflect current thinking.

To facilitate order to the review the subject matter can be classified either by type of constraints and loading or by the optimization method used. The classification based on constraints and loading is as follows:

1. Optimization with static loads and constraints

2. Optimization with dynamic loads and constrains

3. Optimization with aeroelastic requirements

In addition to the three main categories there are papers on probablistic approaches to design, optimization of non-conservative systems, mechanisms and linkages, etc. Another classification could just as well be based on the type of structures. For instance the first two categories can involve trusses, frames, plates, stiffened panels and cylindrical shells or any combination. The second category can also be divided into problems with frequency or response constraints. The third category mostly deals with wings and other lifting surfaces.

The classification based on the optimization method can be as follows:

1. Mathematical programming methods

2. Optimality criteria approaches

3. Methods based on optimal control theory

4. Dynamic programming

The first two methods are widely used in structural optimization while the last two methods are used very little. The mathematical programming methods include linear, nonlinear, geometric and integer programming methods. This classification can be quite nebulous because there can be a great deal of overlapping.

The review papers[15-32] and the proceedings of the important Structural Symposium[33-39] provide a wealth of information on the subject. A brief discussion of the subject matter of the reviews and the proceedings would quickly bring the reader up to date on the available information. The review by Wasiutynski and Brandt[15] traces the history of structural optimization up to the early sixties. It lists a total of 234 papers and covers the period from 1638 to 1962. This paper also contains some thoughts on optimization as related to design and minimum strain energy designs. The comprehensive review by Sheu and Prager[19] covers the period between 1962 to 1968 and contains a list of 340 papers. A brief outline of general methods and applications to

specific problems are part of this review. An extensive review and evaluation of mathematical programming methods as applied to structural optimization was given by Schmit[20]. The survey by Pierson[21] deals with optimization of structures with dynamic constraints. Distinction is made between frequency and response constrained problems and also between parametric and nonparametric behavioral constraints. It was pointed out that the possibility of discontinuous first partial derivatives of the dynamic response is one of the main obstacles to successful application of gradient methods to response constrained problems. The 1972 review by Niordson and Pedersen[22] lists 148 papers. It distinguishes between the optimization of structural elements and structural systems. It also concludes that structural systems can only be dealt with efficiently by mathematical programming methods and structural elements by analytical methods of the calculus of variations. This conclusion is questionable in view of the developments in the last ten years[24,32].

In 1974 Venkayya and Khot[23] made an assessment of the structural optimization programs available in the U.S. The same year Berke and Venkayya[24] reviewed the optimality criteria approaches and pointed out the advantages and shortcomings of these approaches. It was also noted that the computational aspects of optimality criteria approaches are more in tune with the requirements of large scale structural optimization. The importance of the contributions of Prager and his associates to the theoretical development of optimality criteria approaches is well pointed out in this review. An update of the developments in mathematical programming methods for structural optimization was given by Moses[25]. A comprehensive state of the art survey of automated structural design with aeroelastic constraints was presented by Stroud[26]. The paper contains details of theoretical formulations and interpretations of important work in aeroelastic optimization. A review of the applications of optimal control techniques to structural optimization was presented by McIntosh[27]. The consensus is that the application of optimal control theory is limited to simple one dimensional structures. A survey of structural index methods in optimum design was given by Felton[28]. Structural index methods were used extensively in the past in the aircraft industry for component optimization[16,17].

In 1976 Olhoff[29] made a survey of the optimal design of vibrating structural elements. The first part deals with the theoretical formulation and the second with applications. In a very interesting AIAA/ASME SDM lecture, Schmit[30] made a review of the status of various structural optimization methods. Lansing, Lerner and Taylor made an in depth aerospace industry survey of the applications of structural optimization with strength and aeroelastic requirements. This review is particularly interesting because it brings out how little twenty years of research in structural optimization has impacted the actual industry applications. The recent reviews by Kelly, Morris, Bartholomew and Stafford[32] examines

the two important structural optimization techniques (optimality criteria and mathematical programming) and relates them by the use of a Lagrangian function. It also contains recommendations for future software developments in structural optimization.

In addition to the above reviews, there were a number of symposiums and specialty conferences devoted to structural optimization[33-39]. For instance the AGARD symposium on structural optimization[33] contains eleven papers, and they primarily deal with the application of mathematical programming methods. The paper by Schmit deals with an evaluation of the applications of mathematical programming methods. The presentation by Kowalik was on the method of feasible directions. It contained a derivation of the method and a discussion of the implications of the method, but there were no applications to specific problems. The application of linear programming techniques in optimum structural design was discussed by Pope. It contained applications to truss structures and a discussion of the optimization of elastic/perfectly plastic structures. The application of the Sequence of Unconstrained Minimization Technique (SUMT) to the optimization of ship structures was presented by Moe. The method was discussed in the context of preliminary design and contains stiffened panels and frames. The paper by Ashley, McIntosh and Weatherill presented an excellent background for aeroelastic optimization. At the time the state of the art was limited to applications to small problems. Taig discussed the implications of optimization in the preliminary design of aircraft structures. The applications included stiffened panels and cylinders. An interesting discussion of the convergence characteristics of the fully stressed design was given by Melosh in the context of the optimization of pinjointed trusses. Fleming discussed the impact of finite element methods in structural design. Gellatly made an assessment of the problems and the potential in the application of optimization to the design of aircraft structures. The paper by Pedersen discussed the minimum mass layout of trusses. Johnson and Warren discussed the optimization of a supersonic horizontal stabilizer. Optimum thermostructural design of thin circular cylinders was discussed by Vallerani.

Pope and Schmit[34] made an assessment of mathematical programming methods in structural design. Kowalik, Fox, Moses, Ashley, McIntosh, Weatherill and Silver were the other contributors to this evaluation. The AGARD second symposium on structural optimization contains a number of interesting papers. Prager discussed the necessary and sufficient conditions for global optimality. Gellatly made a survey of the state of the art of optimization technology. Venkayya, Khot and Berke derived optimality criteria for generalized stiffness, prescribed displacements, dynamic stiffness and static stability requirements. Their applications included a number of trusses and some frames and plate structures. Moses discussed the status of an optimization program at Case Western University. This program is for the optimization of structural

components such as girders, columns and prestressed concrete beams. It also considers simple gable frames. The structural components can be optimized by SUMT (Sequential Unconstrained Minimization Technique) and structural systems by the method of feasible directions. Templeman, Winterbottom, and Morris presented papers on the applications of geometric programming to structural optimization. Gallagher and Falby discussed the problem of stiffened panel optimization. In an interesting paper, Taig and Kerr applied optimality criteria approaches to optimize truss and plate structures. Taylor and Gwin discussed flutter optimization. Batt and Gellatly's paper dealt with the minimum weight design of surface effect vehicles using a sieve search. In addition the proceedings contain interesting papers: Integer programming (Cella), discriminant function technique (Kiciman and Akgul), shape optimization (Vitiello), the design of compatible structures (Richards), truss configuration optimization (Pedersen), automated sizing by the mixed method (Sobieszczanski and Loendorf), space shuttle design studies (Lansing, Mason, Dwyer and Cooper), A300B aircraft fuselage studies (Schultz), a double iteration method (Finifter) and stiffened shells for radio telescope structures (Kowalewski and Ziebarth).

The IUTAM symposium in Warsaw, Poland in 1973[36] was devoted to structural optimization, and it brought out some interesting papers from Russia and Eastern European countries. Papers on configuration optimization, optimization of frames, trusses and reinforced concrete structures were included in the proceedings. The plastic design of structures was one of the main topics of the symposium. In addition there were papers on probablistic methods and the optimization of shell structures.

The proceedings of the symposium on "Optimization of Structural Design" in Swansea, Wales (Editors: Gallagher and Zienkiewicz)[37] contain important papers on fully stressed design, optimality criteria methods, mathematical programming and reliability based methods. Optimization of trusses, frames, and plate structures were included. The two structural optimization sessions in the ASME Energy Technology Conference (Editor: Venkayya)[38] contained eight papers. The papers dealt with frequency constraints, static displacement constraints and considerations of earthquake resistance designs. Berke and Khot[39] made an interesting study of the convergence characteristics of optimality criteria approaches when applied to truss structures. The aforementioned reviews and the proceedings of the structural optimization symposiums contain almost all the information on structural optimization. The remaining three sections of this paper provide some theoretical background, additional references and some recommendations.

The Optimization Problem and Some Theoretical Considerations

The subject of optimization is a lively topic in almost every discipline. The contributing factor is that optimization, unlike

analysis, is a multivalued problem. In general most optimization problems do not have a unique solution. The usual procedure is to establish a set of necessary and sufficient conditions for an optimum and then construct an algorithm that systematically leads to a solution that satisfies such conditions. If more than one solution satisfies such conditions, the question arises what is the optimum of all such optimums. Since the absolute optimum does not have to satisfy additional conditions, it can be quite allusive in many cases. This uncertainty about the nature of the absolute optimum fostered the development of numerous optimization algorithms. Each of these algorithms may give excellent results for a class of problems. Notwithstanding the enthusiasm of its proponents, no single algorithm can solve all optimization problems with the same degree of efficiency. The evidence of this dilemma is the continued development of new or modifications of the old optimization algorithms. However, in practical design this difficulty is not as disturbing as it sounds. A relative optimum in such cases is generally adequate, because the payoff may not be that significant in seeking an absolute optimum.

In spite of many differences in the methods of solution, most optimization problems can be posed as follows:

Minimize or maximize an objective function which is defined as a function of a design variable vector $\underset{\sim}{A}$,

$$W(\underset{\sim}{A}) = F(A_1, A_2, \ldots, A_m) \tag{1}$$

subject to the constraint conditions that are defined as

$$G_i(\underset{\sim}{A}) = G_i(A_1, A_2, \ldots, A_m) \leq G_{io} \quad i = 1, 2, \ldots, p \tag{2}$$

$$\underset{\sim}{A}^{(L)} \leq \underset{\sim}{A} \leq \underset{\sim}{A}^{(U)} \tag{3}$$

The first set (Eq. 2) may be considered as behavioral constraints and the last set (Eq. 3) as size constraints. Examples of behavioral constraints are stresses, displacements and local and overall stability requirements (strength and stiffness) in the static case, frequencies, stresses and displacements in the dynamic case and flutter velocity and divergence considerations in the aeroelastic case or usually a combination of these. The size constraints are the minimum and maximum limits on the design variables.

The design variables depend on the type of optimization problem. In the design of structural components, such as stiffened panels and cylinders the design variables represent the spacing of the stiffners, the size and shape of the stiffners, and the thickness of the skin. If the skin and or stiffners are made of layered composites, the orienta-

tion of the fibers and their proportions can become additional variables. In the optimization of a structural system (frames, trusses, wings, fuselages, etc.) of fixed configuration the sizes of the elements are design variables. The thicknesses of plates, cross-sectional areas of bars, areas, moments of inertias, and torsional constants of beams represent sizes of the elements. If the optimization includes configuration, the variables must include spatial variables.

The most frequent structural optimization problem is weight minimization. In that case the objective function may be defined as

$$W(\underset{\sim}{A}) = \sum_{i=1}^{m} \rho_i v_i \tag{4}$$

where ρ_i and v_i are the weight density and the volume of the i^{th} structural element, respectively. Unfortunately no explicit expressions can be written for the behavioral constraints of most structural systems.

In a typical mathematical programming application the search for the optimum vector, $\underset{\sim}{A}_{op}$ involves the derivation of an iterative vector of the form

$$\underset{\sim}{A}^{\nu+1} = \underset{\sim}{A}^{\nu} + T^{\nu}\underset{\sim}{D}^{\nu} \tag{5}$$

where $\underset{\sim}{A}^{\nu+1}$ and $\underset{\sim}{A}^{\nu}$ are the design variable vectors in $(\nu+1)$ and ν cycles of iteration. The quantities D^{ν} and T^{ν} represent the direction of travel vector and the step size respectively. In a typical gradient projection method the direction of travel vector is given by

$$\underset{\sim}{D} = \underset{\sim}{D}(\underset{\sim}{f}, \underset{\sim}{g}) \tag{6}$$

where $\underset{\sim}{f}$ and $\underset{\sim}{g}$ are the gradients of the objective and constraint functions with respect to the design variable vector and they are of the form

$$\underset{\sim}{f}^t = \left[\frac{\partial F}{\partial A_1}, \frac{\partial F}{\partial A_2}, \ldots\ldots\ldots, \frac{\partial F}{\partial A_m}\right] \tag{7}$$

$$\underset{\sim}{g} = \begin{bmatrix} \frac{\partial G_1}{\partial A_1} & \ldots\ldots\ldots\ldots & \frac{\partial G_p}{\partial A_1} \\ \vdots & & \vdots \\ \frac{\partial G_1}{\partial A_m} & \ldots\ldots\ldots\ldots & \frac{\partial G_p}{\partial A_m} \end{bmatrix} \tag{8}$$

The search for the optimum vector in mathematical programming methods involves three basic steps.

1. Selection of an initial vector $\underset{\sim}{A}^o$
2. Determination of travel directions
3. Determination of appropriate step size

Judicial selection of the initial vector $\underset{\sim}{A}^o$ is extremely important for obtaining a reasonably good optimum solution. Experience and a few trials can give a good initial vector. The second step involves computation of the constraint and objective function gradients and their transformation to obtain efficient travel directions. This step can be computationally very expensive in some problems and deserves careful examination of the available options. Selection of an appropriate step size is more of an art than a science and often involves numerous trials. The number of iterations and the optimum reached depend on the step size and the initial vector. Reducing the number of iterations is extremely important in large structural optimization problems because each iteration may need one or more analyses and analysis of large structural systems can be very expensive. In the earlier applications of the mathematical programming methods, the number of iterations used to increase in proportion to the number of variables. But recent modifications have alleviated these difficulties considerably.

Applications of mathematical programming and its variations to structural optimization is extensive[40-67]. The reviews by Schmit[20,30], Pope and Schmit[34], Moses[25], Niordson and Pedersen[22] gave a good account of these methods in the past. The number of applications are too numerous to list here. The reviews and the structural optimization symposium proceedings provide all the other references.

Application of optimality criteria approaches involves two distinct steps. The first step consists of the derivation of the necessary optimality criteria for the specified design conditions. The next step is to construct an efficient iterative scheme to achieve these optimality conditions. The usual procedure is to associate the optimality conditions with an appropriate energy or pseudo energy function and use that information in constructing an iterative algorithm. The advantage of this approach is that most analysis procedures can give the information about the energy functions with little additional computational effort. This is expected because the analysis procedures are in general formulated by considerations of the energy of the deformed structure.

As before the optimization problem involves finding the optimum design vector $\underset{\sim}{A}_{op}$ that minimizes or maximizes the objective function as

defined by Eq. 1, subject to the constraint conditions as defined by Eqs. 2 and 3. A new function that combines the objective function and the constraint conditions can be written as

$$\phi(\underset{\sim}{A}) = F(\underset{\sim}{A}) + \sum_{i=1}^{p} \lambda_i \psi_i(\underset{\sim}{A}) \tag{9}$$

where p represents the number of behavioral constraints and the λs are the undetermined Lagrangian multipliers. The constraint functions are defined as

$$\psi_i(\underset{\sim}{A}) = G_i(\underset{\sim}{A}) - G_{io} \leq 0 \tag{10}$$

Minimization (maximization) of the Lagrangian ϕ with respect to the design variable vector $\underset{\sim}{A}$ gives the condition for the stationary value of the objective function with the constraint conditions ψ

$$\frac{\partial \phi}{\partial A_i} = \frac{\partial}{\partial A_i} F(\underset{\sim}{A}) + \sum_{j=1}^{p} \lambda_j \frac{\partial}{\partial A_i} \psi_j(\underset{\sim}{A}) = 0 \tag{11}$$

Simplification of Eq. 11 gives the optimality condition as follows:

$$\sum_{j=1}^{p} e_{ij}\lambda_j = 1 \qquad i = 1, 2, \ldots, m \tag{12}$$

Since there are m design variables there are m such equations, and they can be written in the matrix form as follows:

$$\underset{\sim}{e}\,\underset{\sim}{\lambda} = \underset{\sim}{1} \tag{13}$$

The elements of the matrix $\underset{\sim}{e}$ are given by

$$e_{ij} = \frac{\dfrac{\partial}{\partial A_i} \psi_j(\underset{\sim}{A})}{\dfrac{\partial}{\partial A_i} F(\underset{\sim}{A})} \tag{14}$$

The numerator of Eq. 14 represents the constraint gradients, and the denominator represents the gradients of the objective function. The ratio e_{ij} can be associated with an energy or pseudo energy density in the structure.

The solution of the optimization problem involves (m+p) unknown quantities, where m is the number of design variables and p is the

number of Lagrangian multipliers corresponding to p constraint conditions. However, Eq. 13 represents only m equations. The additional p equations can be obtained by writing the original constraint conditions as follows:

$$\sum_{i=1}^{m} e_{ij} \rho_i v_i = G_{jo} \qquad j = 1, 2, \ldots, p \tag{15}$$

Combining Eqs. 13 and 15 gives the necessary equations for determining the Lagrangian multipliers as follows:

$$\underset{\sim}{H} \underset{\sim}{\lambda} = \underset{\sim}{G}_o \tag{16}$$

where the matrix $\underset{\sim}{H}$ is given by

$$\underset{\sim}{H} = \underset{\sim}{e}^t \, \overline{\underset{\sim}{A}} \, \underset{\sim}{e} \tag{17}$$

$\overline{\underset{\sim}{A}}$ is a diagonal matrix and its i^{th} diagonal element is given by

$$\overline{A}_{ii} = \rho_i v_i \tag{18}$$

The elements of matrix $\underset{\sim}{H}$ cannot be determined explicitly because the $\underset{\sim}{e}$ and $\overline{\underset{\sim}{A}}$ matrices are functions of the design variable vector $\underset{\sim}{A}$ which is itself unknown. Eqs. 13 and 16 are nonlinear sets of equations, and they can be solved only by iterative methods.

Research in structural optimization using optimality criteria approaches involves development of efficient iterative algorithms for determining the design variable vector that satisfies the optimality conditions. It should be remembered that there are no approximations in the development of optimality criteria per se. The approximations are only in constructing efficient iterative algorithms. All iterative methods involving the solution of nonlinear equations tend to be approximate. Mathematical programming methods are no exception. The conclusions that imply that optimality criteria methods are approximate and mathematical programming methods rigorous overlook this distinction. In addition to the difficulty of solving the nonlinear problem, optimization involves the proper handling of active, passive and size constraints. However, there is no need for knowledge of the active and passive constraints of the final design. Knowledge of the constraints of the existing design is all that is necessary for proceeding with the iteration.

Venkayya, Khot and Berke[68-77] have been using optimality criteria approaches for a number of years. They have derived optimality conditions for generalized stiffness, prescribed displacements, static stability requirements and dynamic stiffness[72]. A number of simplifying

assumptions were made in constructing iterative algorithms. For instance specialization of Eq. 16 to a single constraint and some simplifying assumptions give a simple expression for λ as follows for most stiffness type constraints[72].

$$\lambda = \frac{W}{G_o} \tag{19}$$

If the λ's are used simply as weighting parameters, then in multiple constraints their value can be approximated by

$$\lambda_i = \frac{W}{G_{io}} \tag{20}$$

This simplification eliminates the need for the solution of Eq. 16 and the associated positive negative λ's dilemma. Using this approach a number of truss structures were optimized[72]. There are a number of other ways indicated in the literature[78-97].

Optimality criteria approaches are being used extensively for the optimization of a variety of structures[68-110]. They are being extended rapidly to cover many design conditions that were thought to be beyond the scope of these approaches. The next section contains two examples whose optimum designs are obtained by optimality criteria approaches[72].

The references listed so far include review papers, structural optimization symposium proceedings, and papers using mathematical programming and optimality criteria approaches. Some of these papers consider only static cases while others treat both static and dynamic cases. The type of structures optimized are trusses, frames, plates or a combination. The papers dealing primarily with dynamic constraints are listed separately[111-135]. Similarly a few papers dealing with earthquake resistance design are given[136-139]. The literature dealing with aeroelastic optimization is extensive and some notable papers are listed[140-158]. These are followed by papers on frame optimization[159-167]. It should be pointed out that some of the papers listed earlier also discuss frame optimization. The papers on combined mechanical and thermal loading are relatively few and some of them are listed[168-170]. Fully stressed design was one of the earliest attempts at structural optimization and recent studies on its potential and limitations are given[171-179]. Additional papers on applications of Control Theory and other miscellaneous topics are given[188-201].

The list of references given in this paper is by no means complete. Some recent important papers may have been missed inadvertently and not intentionally. However, the papers listed here give a comprehensive coverage of the subject of structural optimization.

Design Example

To give some idea of the present status of the resizing capabilities, a layered composite wing (graphite epoxy) was optimized by two existing computer programs. The programs OPTSTAT[8] and ASOP-3[12] use finite element idealizations for the structure. They both have layered composite and metallic elements. The layered composite material is idealized as an orthotropic material. These programs use only membrane elements for resizing. There are a number of difficulties in mixing membrane and bending elements in the optimization of large structures. For one the local effects of bending tend to dominate and produce impractical material distributions. However, there are successful programs which just optimize frames or plate structures with bending elements. Mixing of membrane and bending elements in the resizing of large aircraft structures has not, at present, been adequately tested. Conceptually, there are no difficulties in optimizing with bending and membrane elements. It is the convergence characteristics of the algorithm and the material distribution that present problems.

A three spar wing shown in Figure 1, is idealized by membrane quadrilaterals, shear panels and bars (axial force members). The top and bottom skins are graphite epoxy layered composite elements with 0°, 90° and ±45° fibers. The spars and ribs are idealized by aluminum shear panels. In addition, the top and bottom nodes are connected by bar elements or posts. The root section of the wing is assumed to be fixed. The wing is designed for two independent loads. These loading conditions are generated by simplified pressure distributions representative of a subsonic, forward-center-of-pressure loading and a supersonic near-uniform-pressure loading. The detailed distribution of the loading on the nodes is given in Table 1. The material properties of the graphite epoxy and aluminum are given in Table 2. The constraints are only on stresses and minimum sizes. The wing was optimized by OPTSTAT and ASOP-3. The distribution of the composite layers and the thickness of the spars and webs are given in Figure 2.

The final weight of the wing obtained by the two programs is not significantly different, but the difference in material distribution is more pronounced. These differences can be attributed to the way the two programs resize their elements. The OPTSTAT program resizes the elements by using an energy criterion, while ASOP-3 resizes by the fully stressed design criterion. The implications of the differences in material distribution has yet to be investigated.

Conclusions and Recommendations

Structural optimization algorithms have been continuously improved over the past twenty years. The mathematical programming and optimality criteria methods are emerging as practical and viable approaches to large scale structural optimization. As the efficiency of the mathematical programming is improving, the optimality criteria are being extended to include more and more complex design conditions. As the understanding of the behavior of these approaches improves, it is becoming increasingly evident that they are not that different except in interpretation. In actual implementation the differences are less significant. Even though the present consensus is that the methods based on optimal control theory can be used only for small component optimization, there may be a possibility of improving them to compete with the other two approaches.

The number of theoretical investigations and applications to small problems is increasing. However, extension of optimization to large practical problems is slow because of the uncertainties of the convergence characteristics and the large number of design conditions that have to be included in a practical design. It is extremely important that the designer include as many conditions as necessary even if some of these appear to be insignificant initially. The old practice of designing for a few critical conditions can be unconservative in optimization. The neglected design conditions can have a significant effect on the optimized structure. Moreover it is a well known fact that optimized structures can be grossly non-optimal for conditions other than those for which they are designed. As more and more conditions are included, the shear volume of information to be manipulated becomes vast and unwieldy for efficient programming. In addition the convergence characteristics and the problems of relative minima become increasingly difficult in the case of large problems. Because of these difficulties there is apprehension and some resistance to incorporation of sophisticated and theoretically sound optimization algorithms in large programs. At present the designers are content with simple algorithms like fully stressed design. However, such over simplifications can be counterproductive in complex optimization situations.

A more disturbing fact is that much of the current research in optimization is concerned with theoretical aspects like algorithm suitability, convergence characteristics and economy of computations. There is little attempt to verify the validity of the optimized designs in actual service conditions. The rationale for this neglect is that the analysis procedures in optimization are already verified and any further verification simply does not add new knowledge. This argument is not valid simply because optimized structures tend to be more sensitive to fabrication defects and improper definition of the loading environment. Service life testing of optimized structures can provide

valuable information for defining adequate requirements for the optimization of practical structures.

Another important area of concern is the incorporation of the latest damage tolerance, fatigue life and fracture mechanics requirements into the optimization scheme. This involves translating these requirements into concise and coherent stress, displacement and other stiffness type constraint data. If these requirements are not included in the optimization scheme, later modifications for meeting them can be counterproductive. In addition it is worthwhile investigating the possibility of incorporating hypothetical damage models into optimization. Inclusion of such models can reduce the damage sensitivity by providing adequate redundancy around the damage. If the damage is interpreted as a loss of stiffness, the analysis may not become too complicated. Specification of all these requirements sounds quite ominous but the resulting benefits can more than compensate for the effort by producing reliable, efficient and safe structures.

Biographical Sketch

Dr. Vipperla B. Venkayya was born in India on May 16, 1931. He received his undergraduate education in India. He came to the United States in September 1957 to pursue graduate studies. He received an an M.S. from the University of Missouri in Columbia in 1959, and Ph.D from the University of Illinois in 1962.

In February 1962 he joined the faculty of the School of Engineering, State University of New York at Buffalo, as Assistant Professor. He taught graduate and undergraduate courses in Structural Analysis, Design, Structural Dynamics and Numerical Methods. He joined the Air Force Flight Dynamics in 1967 as an Aerospace Engineer in the Structural Mechanics Division and engaged in research and development of structural analysis and optimization methods. He is an Adjunct Professor of Aerospace Engineering in the Air Force Institute of Technology. He was recently appointed as Associate Editor of the AIAA Journal in the area of structural analysis, optimization and composite structures. He is a member of American Institute of Aeronautics and Astronautics, American Society of Civil Engineers, and the American Society of Mechanical Engineers.

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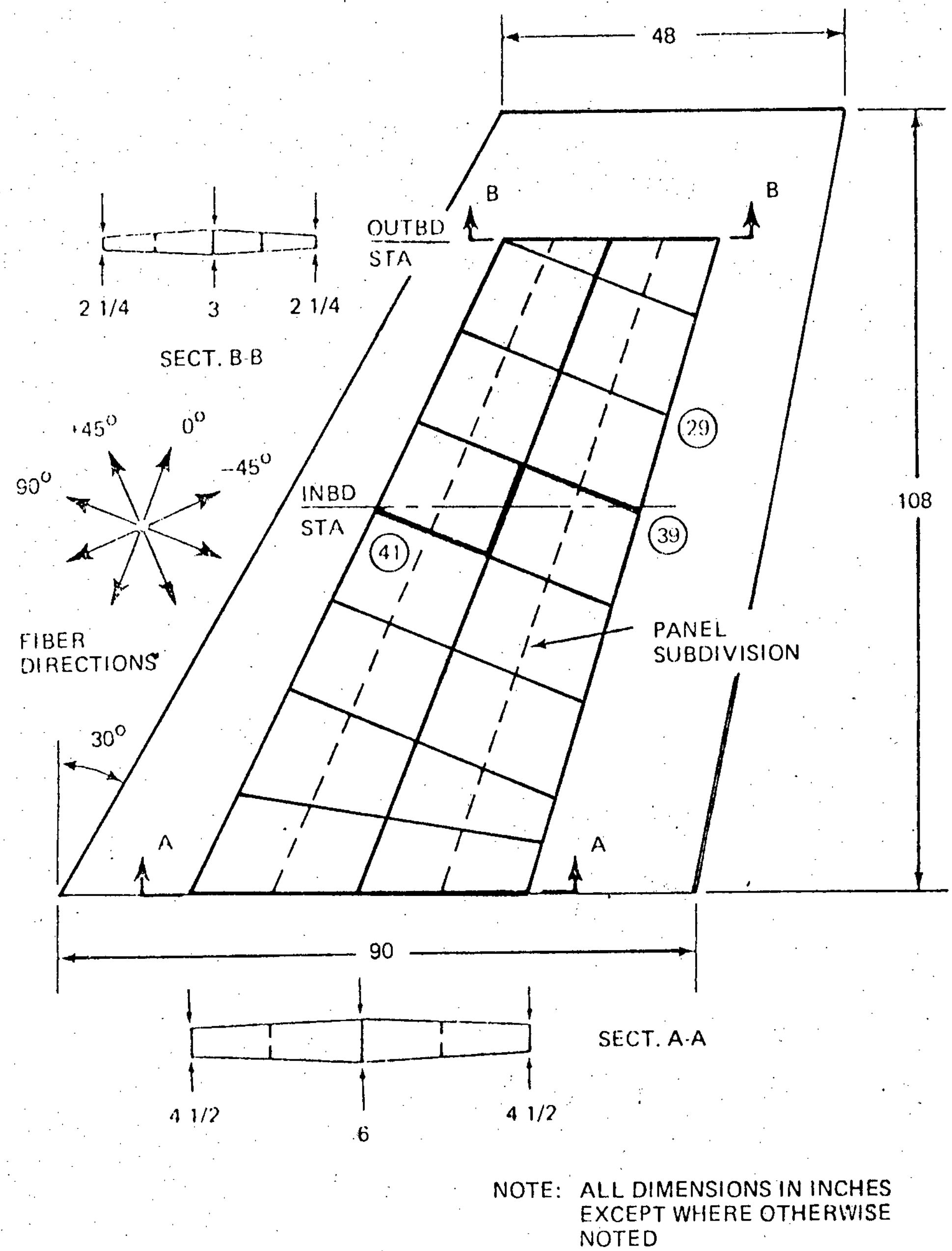

Figure 1. Aerodynamic Planform and Primary Structural Arrangement of Wing

Note: Top figures are from Program OPTSTAT; bottom figures ( ) are from Program ASOP 3.

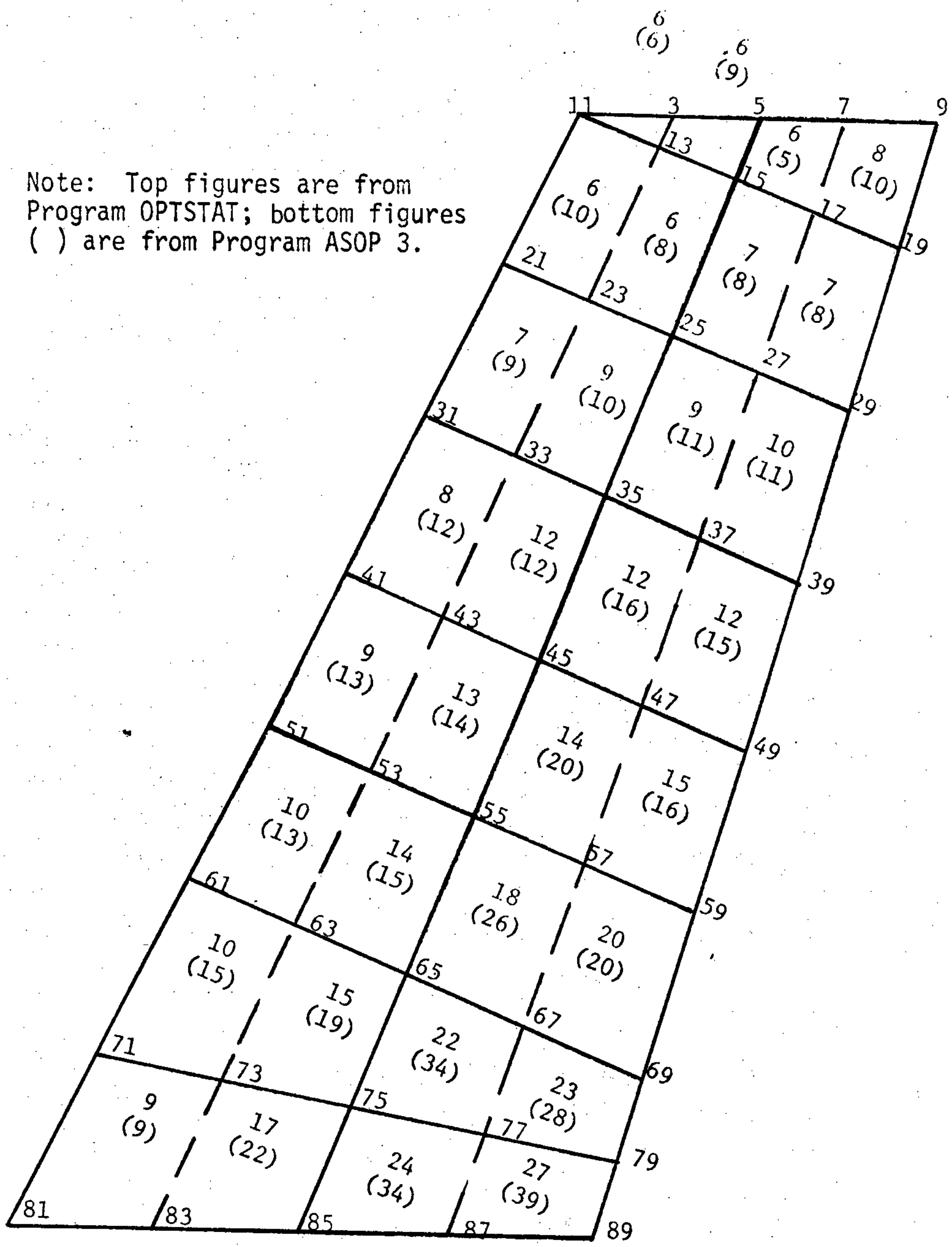

Figure 2a. Total Number of Layers in the Top Skin
(Bottom skin is the same)

Note: Top figures are from Program OPTSTAT; bottom figures ( ) are from Program ASOP 3.

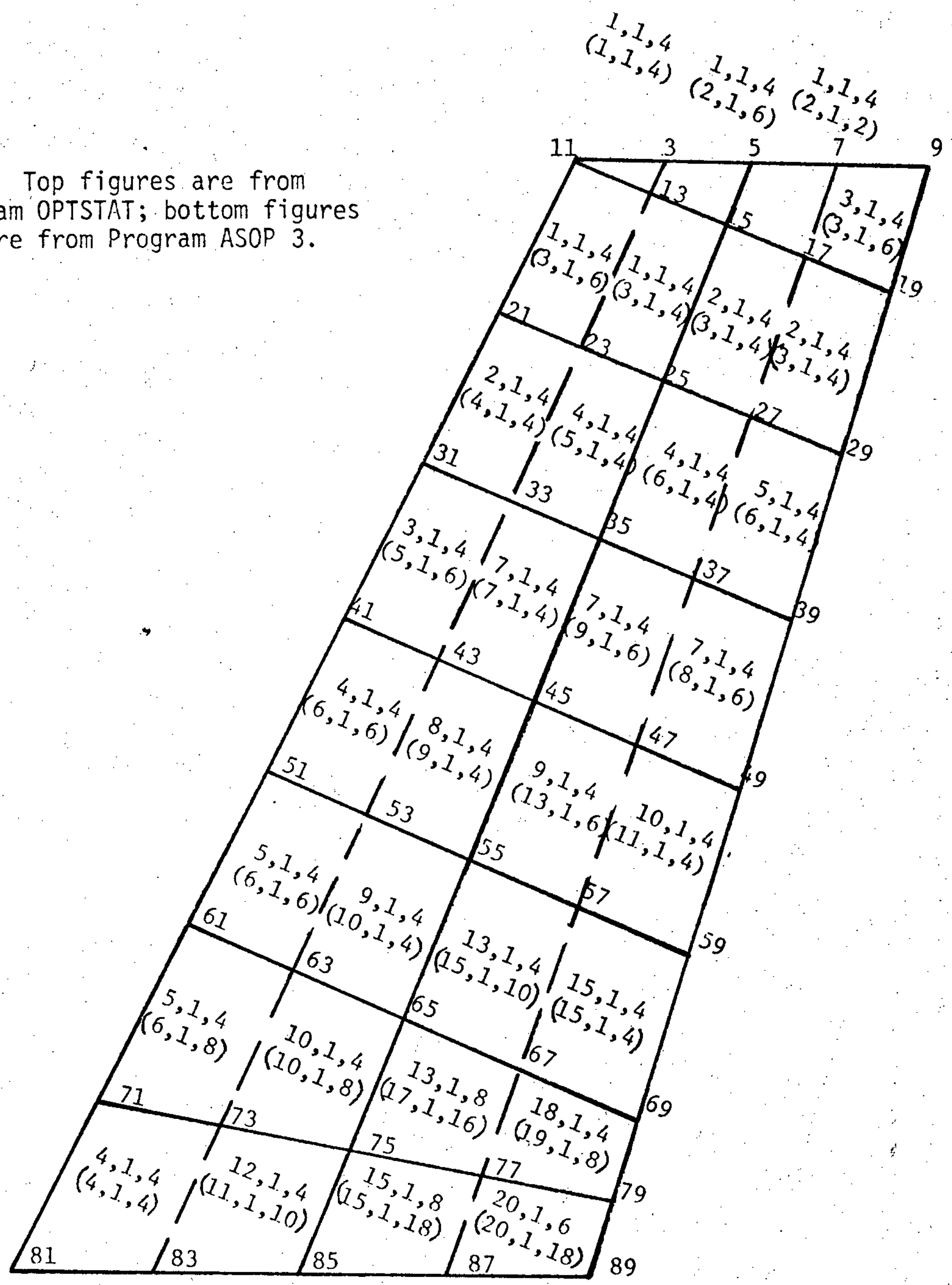

Figure 2b: Distribution of Fibers in 0°, 90°, ±45° Direction

Table 1a. Loading Data

Node No.	x L.C.1	x L.C.2	y L.C.1	y L.C.2	z L.C.1	z L.C.2	Node No.	x L.C.1	x L.C.2	y L.C.1	y L.C.2	z L.C.1	z L.C.2
1	0	0	0	0	29	30	35	0	0	0	0	365	230
2	0	0	0	0	29	30	36	0	0	0	0	365	230
3	-2800	-2420	-6960	-6020	1130	979	37	-4250	-1270	1740	518	974	336
4	2800	2420	6960	6020	1130	979	38	4250	1270	-1740	-518	974	336
5	0	0	0	0	91	56	39	1820	4160	-743	-1700	694	1410
6	0	0	0	0	91	56	40	-1820	-4160	743	1700	694	1410
7	-9870	-4020	-9780	-3980	1130	474	41	0	0	0	0	365	402
8	9870	4020	9780	3980	1130	474	42	0	0	0	0	365	402
9	205	351	-7380	-12600	926	1530	43	0	0	0	0	378	313
10	-205	-351	7380	12600	926	1530	44	0	0	0	0	378	313
11	0	0	0	0	178	194	45	0	0	0	0	392	247
12	0	0	0	0	178	194	46	0	0	0	0	392	247
13	0	0	0	0	214	175	47	-4440	-1320	1820	541	1050	361
14	0	0	0	0	214	175	48	4440	1320	-1820	-541	1050	361
15	0	0	0	0	253	157	49	1890	4330	-773	-1770	742	1500
16	0	0	0	0	253	157	50	-1890	-4330	773	1770	742	1500
17	-5680	-1600	2320	653	1020	325	51	0	0	0	0	390	430
18	5680	1600	-2320	-653	1020	325	52	0	0	0	0	390	430
19	2310	5510	-946	-2250	723	1550	53	0	0	0	0	404	334
20	-2310	-5510	946	2250	723	1550	54	0	0	0	0	404	334
21	0	0	0	0	314	347	55	0	0	0	0	420	264
22	0	0	0	0	314	347	56	0	0	0	0	420	264
23	0	0	0	0	326	270	57	-4640	-1380	1900	565	1120	386
24	0	0	0	0	326	270	58	4640	1380	-1900	-565	1120	386
25	0	0	0	0	338	213	59	2290	5300	-937	-2170	883	1820
26	0	0	0	0	338	213	60	-2290	-5300	937	2170	883	1820
27	-4070	-1210	1660	496	902	311	61	0	0	0	0	413	458
28	4070	1210	-1660	-496	902	311	62	0	0	0	0	413	458
29	1740	3990	-713	-1630	646	1310	63	0	0	0	0	391	326
30	-1740	-3990	713	1630	646	1310	64	0	0	0	0	391	326
31	0	0	0	0	340	375	65	0	0	0	0	368	233
32	0	0	0	0	340	375	66	0	0	0	0	368	233
33	0	0	0	0	352	291	67	-3030	-922	1240	377	804	287
34	0	0	0	0	352	291	68	3030	922	-1240	-377	804	287

Node No.	x		y		z		Node No.	x		y		z	
	L.C.1	L.C.2	L.C.1	L.C.2	L.C.1	L.C.2		L.C.1	L.C.2	L.C.1	L.C.2	L.C.1	L.C.2
69	3070	7160	-520	-1210	1040	2180	74	0	0	0	0	370	310
70	-3070	-7160	520	1210	1040	2180	75	0	0	0	0	304	194
71	0	0	0	0	433	484	76	0	0	0	0	304	194
72	0	0	0	0	433	484	77	-1370	-451	262	86	446	175
73	0	0	0	0	370	310	78	1370	451	-262	-86	446	175

Note: Node (i) in Table 1 corresponds to Node (i + 2) in Figure 2.

Table 2. Material Properties

Graphite Epoxy:

Density = 0.055 lbs/in^3

Layer Thickness = 0.005 in.

Allowable Stresses = 115,000 psi

Elastic Constants

$E_{11} = 18.5 \times 10^6$ psi

$E_{22} = 1.6 \times 10^6$ psi

$G = 0.65 \times 10^6$ psi

$\nu = 0.25$

Aluminum:

Density = 0.1 lbs/in^3

$E = 10.5 \times 10^6$ psi

$\nu = 0.3$

USE OF FULL MISSION SIMULATION FOR AIRCRAFT SYSTEMS EVALUATION

BY

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Use of Full Mission Simulation for Aircraft System Evaluation

Abstract

Full mission simulation to evaluate aircraft subsystems gives the user the capability of testing the given system in all portions of the flight envelope. This becomes important when a critical issue associated with the system is the workload levels that develop in using the equipment. The Total Aircrew Workload Study (TAWS) simulation was designed to evaluate a new avionics concept in an advanced assault transport aircraft with a reduced flight deck crew. A key issue, once it was determined that the avionics design could perform the integrated communication/navigation task, was to verify that the system was usable in a real world environment. Hypothetical mission profiles were designed and flown in simulation to study the workload present in each phase of the tactical mission. Crews currently flying Air Force transport missions were used to evaluate the simulation. The full mission capability permits a more complete investigation of the equipment usage to be made and allows for more versatility in the mission profiles. While not necessary in all simulation applications, for some evaluations the full mission capability could prove invaluable.

Introduction

The Total Aircrew Workload Study (TAWS) simulation was conducted at the hybrid computer facility in the Control Synthesis Branch of the Air Force Flight Dynamics Laboratory at Wright-Patterson AFB, Ohio (AFFDL/FGD). The simulation programming and operation ran from May 1976 through November 1977. Thirteen months were devoted to extensive software/hardware development and system integration. One month was used in system validation, with the remaining five months devoted to data collection.

The program was a joint effort with the Flight Dynamics Crew System Integration Branch (AFFDL/FGR). Much of the human factors engineering and system requirements were developed by FGR, with the software development and hardware implementation being done by FGD.

The TAWS simulation was conducted in response to a request from the Military Airlift Command (MAC) and the Advanced Medium STOL Transport System Program Office (AMST SPO). It had been proposed that one of the flight deck crew members, the navigator, could be deleted from the crew to reduce life cycle cost. It became necessary to determine the level of avionics required in the cockpit to permit the pilot/copilot team to accomplish the navigation task as well as the flying and communication tasks. Once an avionics package was defined that could accomplish the required tasks it had to be evaluated in a mission context to determine its usefulness. Even though the package could successfully handle the navigation task, if it created a high workload situation it could jeopardize rather than support the mission effort. Workload levels in all areas of the mission profile needed to be evaluated.

The TAWS simulation was the final phase in a three-phase study to determine cockpit requirements for an advanced tactical aircraft. Mission analysis and mock-up studies had preceded simulation. Mission content was defined by a hypothetical, 1980's tactical assault scenario. An advanced cockpit design, including a fully integrated avionics suite, was developed in accordance with the mission requirements.

The simulation was used as a testbed to evaluate both the advanced avionics concept and the possibility of deleting one crew member from the three-man transport flight deck. A primary design requirement for the simulation was to provide full-mission capabilities in a man-in-the-loop simulation. Thus the crew workload could be evaluated in a nearly "real world" situation.

The simulation used a generic STOL (Short Takeoff and Landing), six degree of freedom hybrid aircraft model. An integrated communication/navigation

system provided global navigation abilities with an active interface between the crew and the Nav computer. An auto-pilot/flight director model was integrated with the aircraft control system to provide fully automated flight. This system could be coupled with the navigation system for automatic, three dimensional flight profiles.

Current flight crews participated in the evaluation and data collecting phase of the simulation. Preliminary findings indicated that the avionics package could perform the mission tasks, although workload did approach unacceptable levels for some tasks. The evaluation also produced suggestions for simplifying the man/machine interface to improve system utility.

"The objective of the Total Aircrew Workload Study is to evaluate/validate the crew station design requirements for an advanced medium STOL transport (AMST) with a two pilot/one loadmaster crew." (TAWS test plan, Page 1. BR-WP#4906-604-6-Final Revision, 30 August 1978).

To meet this objective and to answer the questions raised by MAC and the AMST SPO it was determined that full mission simulation was mandatory. The mission scenario used was a hypothetical tactical assault mission. An extensive mission analysis was undertaken to project into the future and to predict mission and fleet requirements. As a result a tactical assault mission scenario was developed. From the mission scenario alone several simulation requirements are apparent. The ability to simulate:

(1) Realistic take-off, landing and ground operations associated with troop/cargo pick-up and drops

(2) Ground operations to prepared/unprepared and conventional/short (or STOL) airstrips

(3) Air drops - both high altitude cargo drops and lapes (low altitude parachute extraction system) maneuvers; also in-flight troop drops

(4) Diversions

(5) Full weather (IMC/VMC-Instrument and visual meteorological conditions) and day/night flying capabilities.

(6) Low and high altitude flying

(7) Formation flight

(8) Threat environment

are required by the mission definition.

The avionics suite proposed from mission analysis and mock-up studies and other full mission considerations imposed more requirements for the TAWS simulation and can be broken into four distinct areas. These four requirements were:

(1) The aircraft model

(2) A global navigation system

(3) The emulation of additional mission workload

(4) Hardware fabrication.

The aircraft model simulated needed to have performance capabilities and handling qualities similar to those of the AMST prototype aircraft. There was no requirement for a specific aircraft, therefore a generic STOL (short take-off and landing) model was simulated. The basic aerodynamic package used was that of the McDonnell-Douglas YC-15 AMST prototype. An inhouse simulation of this aircraft had been done earlier, thus it was a straight-forward task to bring up the aircraft model again. In order to make the aircraft model a generic one, several modifications were made to both the aero model and the flight control system.

Major changes to the free airframe aero model included (1) the addition of a drag term as a function of Mach number, (2) complete re-programming of the spoiler actuation model into a more simple system, and (3) adding an incremental change to C_m due to alpha. The original flight control system was augmented by the addition of a flight director system and a two-axis autopilot. Gain scheduling in the original data package was modified to enhance cruise condition performance. System gains were changed in several loops of the control system when the new autopilot commands were interfaced.

The changes outlined above served to mask or alter the flying qualities of the original AMST. The aircraft then was truly a generic model and as such should not be identified as either prototype (or any aircraft in the Air Force inventory). These many changes served to fine tune the flying capabilities and handling qualities of the simulation to the TAWS requirements. Thus the need to simulate a new airplane was avoided.

The original data package used was a 1974 package. The equations and data were from a company proprietary report, therefore no detailed description of the data package will be given. General system descriptions, where appropriate, will be discussed in other sections of this report.

The second requirement area was that of a global navigation system. The inertial navigation system (INS) concept tested in simulation was modeled after the Collins ANS-70 INS system. The functions which the navigation computer was to perform were patterned from this operational system. Key areas of difference were in specialized functions, display options, and modes of pilot/computer interactions.

The basic design requirements for this prototype avionics suite were five-fold. Bulk data storage for named way points along a flight plan, in addition to the capability to define latitude/longitudes, bearing/distance, and bearing/bearing coordinates from known points, was desired (i.e. the Nav computer could recognize a 3 or 4 letter identifier as a specific earth coordinate with its associated ratio frequencies). Direct pilot interaction with the Nav Computer for interrogation and updating capabilities was required. A cathode ray tube for display of navigational information was required. Specialized LED (Light Emitting Diodes) readouts were fabricated for communications, navigation and aircraft parameters readouts. An automatic airdrop calculation was a required function. It was required that the navigation computer could be coupled with the flight director/autopilot/flight control systems for fully automatic three dimensional navigation.

The emulation of other phases of mission workload was done to enhance simulation realism. However, it served a more critical role in the over-all evaluation of the avionics system. Without the simulation of these other real world tasks, much of the evaluation of the system would have been purely conjecture. Accurate workload measures could not have been made. Therefore it was deemed necessary to include in simulation other tasks critical to the tactical mission in addition to the flying of the aircraft.

The flight deck crew interaction with the loadmaster was simulated by actually having a loadmaster fly the practice and data missions in a mockup of the loadmaster's compartment. Interaction with other members of the flight formation was also simulated. Station keeping equipment (SKE) was installed in the simulator and the crews were tasked to perform SKE tasks as they flew in formation. (The simulation called for them to fly as lead ship in both a 3-ship and a 9-ship flight and also as a lone ship.)

The most demanding task in the cockpit environment is the communication function. A full AIC-18 communication network was used in simulation. The crew was required to carry out all normal tactical communications within their ship, with other members of the formation, with ground control, troops, and support aircraft.

Additional mission realism and workload were created by the actual mission scenarios used. Initial training flights were flown in the Delaware area of the United States. Additional training flights and all data missions were flown in West Germany along the German border. This required the crew to be in compliance with buffer zone regulations, to exercise "see-and-avoid" tactics, and to use the appropriate counter measures in the threat environment. The role

playing required here was helped by the experimenters who handled the communications task.

The last additional task imposed on the crew was not one normally found in the tactical mission. To have some measure of the workload level present during various mission phases, a secondary task was introduced. Literature has shown that workload can be measured by evaluating the performance of some secondary task. The task chosen was a time estimation task.

Baseline estimates of a 10 second time period were made for both pilot and copilot. These estimates were taken when the crew had no other tasks to perform. Once an individual's baseline was established in a no workload environment, the same time estimation task was performed in the mission environment. Crew members were cued by an audible tone during various segments of the mission. On hearing the tone, the crew member started his own timer. He stopped it after he thought ten seconds had elapsed. Estimates were taken during critical mission segments such as the approach to a drop zone, flight through the drop zone and intervals of high communication.

As the workload increased the pilot time estimates diverged from baseline. The period of time generally increased. At times, if workload was very intense, the estimation task was started and forgotten or ignored entirely.

The hardware fabrication and installation required for TAWS was a major part of the simulation effort. The program requirements called for equipment which were currently not "off-the-shelf" items. Requirements also dictated the reconfiguration of the simulator cab to meet the proposed cockpit design. Major hardware pieces that were fabricated included the:

(1) Integrated Communications/Navigation (ICN) panel

(2) LED readouts for special navigation information and aircraft parameter displays

(3) Interactive keyboard for the Nav system

(4) Flight control system/autopilot control panels

(5) Communications system built from AIC-18 units

Cockpit reconfiguration included the installation of some new instruments, moving of instruments to allow for the placement of special displays (LED's,

CRT, keyboard) and extensive modification of the center console (aft of the throttle quadrant) to allow for the installation of the ICN panel.

The navigation system simulation was the focal point of the TAWS effort. The study was conducted for the purpose of evaluating an advanced avionics concept. The navigation management system was the key element within this concept. Advanced integration of other systems (e.g. autopilot, communications, advanced LED and CRT displays) with the Nav system provided greater flexibility in the cockpit.

The basic concept for the Nav management system model was taken from the Collins ANS-70 inertial navigation system. Specific system definition of the desired functions, formats and software algorithms were defined in-house. The Collins concept was heavily modified to fit the tactical military mission. The manner in which the navigation system interfaced with the other aircraft systems was also designed to best meet the mission requirements.

The system simulated has a multiple function capability. It provides bulk data storage, navigation aids information,control system steering information, extensive display capability and does all the necessary navigation calculations. In addition to this the system is designed with a keyboard/CRT system which allows the crew to interact actively with the system. One special military function designed into the system called for the Nav management system to store all of the airdrop information and perform the required drop calculations.

The bulk data storage function permitted the storage of three or four character identifiers for waypoints of interest. The named waypoints with information concerning latitude, longitude and elevation, and navigational aids available could be loaded into the system from data cards prior to system use. The bulk data storage increased system flexibility and also decreased pilot workload. Waypoints could be specified by name rather than by latitude/ longitude coordinates.

A baseline configuration was also available in the system. With the baseline system there was no storage capability. Waypoints could not be defined by their letter identifiers. Use of the baseline Nav computer required that all Nav locations be defined by the latitude and longitude of the point. This input format required more work on the part of the crew to enter or modify information in the system. Once waypoints were input to the system all other data manipulations and calculations were the same for the baseline system as they were for the bulk storage system. Waypoints in the baseline system had generic names generated by the Nav computer. All had a form of LLXX (XX being a

two digit number) which was another inconvenience of the system. Crew members had to note what lat/long in the system defined a given point in the flight plan.

The baseline system, although not as convenient to use, had the capability to perform the required task. By being a more austere design it would be less costly to implement. Thus it was evaluated to determine if it would be a cost effective alternative.

Thus the bulk data storage simplified system operation. The system was also capable of accepting inputs in three other formats. Waypoints not having identifiers stored in memory could be specified by latitude/longitude, by giving a bearing and distance from a defined waypoint, or by specifying it as the intersection of bearings from two defined waypoints. The navigation management system automatically generated character names for such inputs in the form of LLXX, BDXX and BBXX. The generic names come from lat/long (LLXX), bearing/distance (BDXX) and bearing/bearing (BBXX) definition. The XX is a two digit number. Points so defined are numbered sequentially as they are defined. Both bulk and baseline systems accept these input formats.

The system stores information on available navigation aids in the area of mission operation. This area is available to the crew for navigational use. It also stores communications information - both emergency frequencies and available VOR, ADF or TACAN frequencies and channels at a waypoint. The nav system uses this information to generate glideslope and localizer drive signals on ILS (Instrument Landing System) and MLS (Microwave Landing System) approaches.

The system is designed to be integrated with the flight control system of the aircraft. Flight plan information is set up to permit fully automatic three-dimensional navigation. Assuming a reliable and precise autopilot/FCS design, this frees the pilot to devote more time to other required cockpit functions.

The system was designed with an extensive display capability. Several specialized LED displays for comm/nav and aircraft parameters were designed into it. With the CRT/keyboard system the crew could interrogate the navigation management system and determine the mission status. The CRT display capability made eight separate pages of information available to the crew. These pages included (1) the flight plan page, for flight plan programming, (2) the nav aids page, listing stored navigational aids, (3) the nav aid data page, giving additional information such as lat/long, elevation and available

frequencies and channels for the navigation aids lists on the page, (4) the emergency data page, which listed necessary information for quick access in emergency conditions, (5) the airdrop page, which contained the parameters used to define a drop zone, (6) the waypoint data page, which gave information defining the aircraft position relative to a specified waypoint, (7) the progress page, which presented information concerning progress relative to the flight plan and also had the control system inputs for vertical velocity commands, and (8) the present position page, which displayed current aircraft parameters as well as current geographic location.

The system also performed all of the necessary navigation calculations. The calculations to determine bearings, distances, etc. were modeled using spherical earth methods. The four analogies of Napier and Gauss' Formulas were used to solve for the desired variables in the terrestrial oblique spherical triangle. Magnetic variance for the Dover and Germany areas were also simulated using a first order approximation to determine the constants.

One of the most critical points in the design of the system was the design of the actual man/machine interface. The system had to be easily accessible to the crew for it to be useful. There were two methods available to input information to the system. One method was a manual tuning method which used knobs to dial in desired frequencies. The other technique was an interactive keyboard which was the primary input device to the navigation system.

The tuning knobs were used to input radio frequencies to the system. The intent of the knob input was to provide a backup capability should the keyboard functions be lost in flight. Since this input function was limited it was designed into the cockpit as a "get-home" capability. It was not intended for use in completing the tactical phase of the mission, but rather to insure the crew had a backup to get safely to a home base should the primary system fail.

An important function of the navigation system was the calculation of the computed air release point (CARP) for the drop sequences. All the necessary data for these calculations was entered on the airdrop page. The necessary coordinate, drop attitudes, temperatures, winds, etc., were input to the system. The nav management system could use all this data to calculate the proper point for drop release such that the dropped cargo would fall within the specified drop zone.

The simulation was more concerned in evaluating the procedural technique of using the nav system in this manner than in fully simulating the drop calculations with all the ballistics-type equations. The actual CARP calculations

were greatly simplified. The CARP was defined as a rectangular area adjacent to the projected impact point (PIP). If the load were released any time while the aircraft was within the confines of the area, the drop would be good. Simple geometry was used to check the aircraft position versus the CARP zone and determine when the aircraft was in the zone.

The autopilot/flight director/flight control system was developed in such a fashion that it could be integrated with the navigation system to perform three dimensional navigation. An autopilot and flight director system were designed in-house to meet the design requirements. The original AMST flight control system was modified to accept flight director commands and to integrate with the autopilot. Multiple mode configurations were available for greater flexibility of the integrated options.

When integrated with the navigation system, the flight control/autopilot system was capable of performing multiple functions. The most sophisticated of these was a fully automatic, three-dimensional navigation route around the specified flight plan. By proper mode selection, variations to the flight plan could be implemented. Vertical profile commands could be given to fly a given flight path angle or a given vertical velocity. The system was designed to give shallow pitchovers not to exceed 0.05g.

The lateral modes also had several available functions. Course capture could be effected by flying either a heading hold or a course hold mode. Once on course a desired ground track was maintained. The lateral autopilot thus held the required crab angle to offset any crosswinds present. Commanded bank angles were limited to 30^{o} bank as a maximum roll angle to avoid overstressing the aircraft.

The flight director system could also be uncoupled from the autopilot. In this configuration the flight director bars displayed the commanded information from the nav system which the pilot could then manually fly.

Once validated the simulation was used to evaluate the avionics system. Flight schedules included one day of traffic patterns and landings/takeoffs for aircraft familiarization, two days of practice flights for navigation equipment training and one to two days of data taking.

The data mission used included a take off and a low altitude drop of equipment in Germany. There were four replications of the run using the following configurations:

(1) Auto-pilot off; Bulk storage in nav computer

(2) Auto-pilot off; Baseline storage

(3) Auto-pilot on; Bulk storage

(4) Auto-pilot on; Baseline storage

Order of flights was varied with each crew so that it was not a factor in the data. Subjective data was taken by debriefing the crews at the completion of the data taking. Objective data measuring course, altitude and airspeed deviation was recorded. A statistical analysis was performed on these variables to determine the statistical differences among the various configurations.

The TAWS simulation was not only a workload study to determine the feasibility of an advanced cockpit design for a two-man flight deck crew. It was also a study to determine the feasibility of doing full mission simulation as opposed to simulating only a phase of the mission profile. Many conclusions about, suggestions for, and possible improvements to the avionics suite resulted from this simulation and will impact further efforts in this area.

Integrating all the subsystems required by the project description produced a complex simulation. The several new technologies were integrated to produce a realistic full-mission environment. Full-mission simulation is a viable alternative in the simulation field and is an attractive alternative for some Air Force applications.

There are some guidelines to use in setting up full-mission tasks. The first is to determine if a simulation of that magnitude and scope is really needed. Increasing system complexity just to have a capability that probably won't be used only increases risk of down time. Timely and accurate system definition is essential to timely programming. In a full-mission simulation any components that can be simplified should be. For instance, in the TAWS simulation the aircraft model could have been far simpler than the one used. A good method for integrating the many technologies and for troubleshooting the system needs to be developed.

There are applications both in research areas and training area which may require full-mission technology. Some tasks, such as the TAWS navigation task, cannot be adequately investigated during a short mission segment. Over-all mission integrity must be maintained to insure valid system usage. This may become even more true in the area of cockpit design and optimization.

Workload measures will be an important criteria in the acceptance or rejection of a cockpit configuration. This workload measurement must be accurate to be of any value. To be accurate the mission environment will need to be simulated. The complexity required must be determined for the individual program.

Many conclusions, with respect to both avionics design and the full mission simulation concept, were drawn from the program. The avionics package contained all of the design features necessary to accomplish the tactical mission. Analysis of the data indicated that the four configurations tested showed little or no statistical difference in the objective tracking data. The primary difference among these configurations was noted in the workload measurements and the subjective data. While the crews could accomplish the mission with each system, the workload increased tremendously as the system became more austere.

Even with the most sophisticated system, pilot comments indicated a great deal of fine-tuning was necessary. The workload in the cockpit was not allocated evenly, with the copilot absorbing most of the navigator's functions in addition to his own. As he became more involved with the communication and navigation tasks, the copilot was essentially lost as a safety man in the right seat. One of the biggest questions was that of over-all system safety with the reduced flight deck crew size. Essential tasks such as "see-and-avoid" measures, instrument monitoring, emergency conditions and others may be extremely difficult to handle in a two-man cockpit.

Refinements to the avionics design to balance the workload for pilot/copilot would alleviate some of this problem. Another suggestion was to keep a third crew member in the cockpit. This member need not be a navigator but could be an enlisted person in the role of a systems operator. This crew member's function would be to handle checklists, communications and provide the all-important extra pair of eyes for monitoring in and out of the cockpit. Further studies in this area are anticipated to answer these and other questions.

More conclusive results could be drawn in regard to the simulation itself. Though previously not attempted, the full-mission simulation with all its complexities was a successful effort. A detailed methodology for integrating all of the subsystems of the program as well as a technique to validate the total system was developed. This methodology is generalized enough to be applicable to follow-on programs of a similar magnitude. A full-mission simulation, if judiciously planned and executed, can permit an in-depth study of a major aircraft subsystem. All phases of a flight profile could be tested without having to

turn to expensive flight tests, in the case of existing hardware, or to the fabrication of test equipment in the case of a developmental program.

Biographical Sketch

Karyl A. Adams was born in Urbana, Ohio on January 8, 1952. She graduated from Ohio Northern University in 1975 with a Bachelor of Science Degree in Electrical Engineering.

In January 1976 she began working in the Air Force Flight Dynamics Laboratory, Control Synthesis Branch as an electronics engineer doing hybrid aircraft simulation. Simulation efforts include the XQM-103 remotely piloted vehicle study, Total Aircrew Workload Study (TAWS) and the YF-16 CCV (Controlled Configured Vehicle) study. Job responsibility is to develop, program and verify analog and digital software for hybrid aircraft simulation. The author served as the project engineer for the TAWS simulation team.